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Journal

of

THE ROYAL AERONAUTICAL SOCIETY

JANUARY 1958

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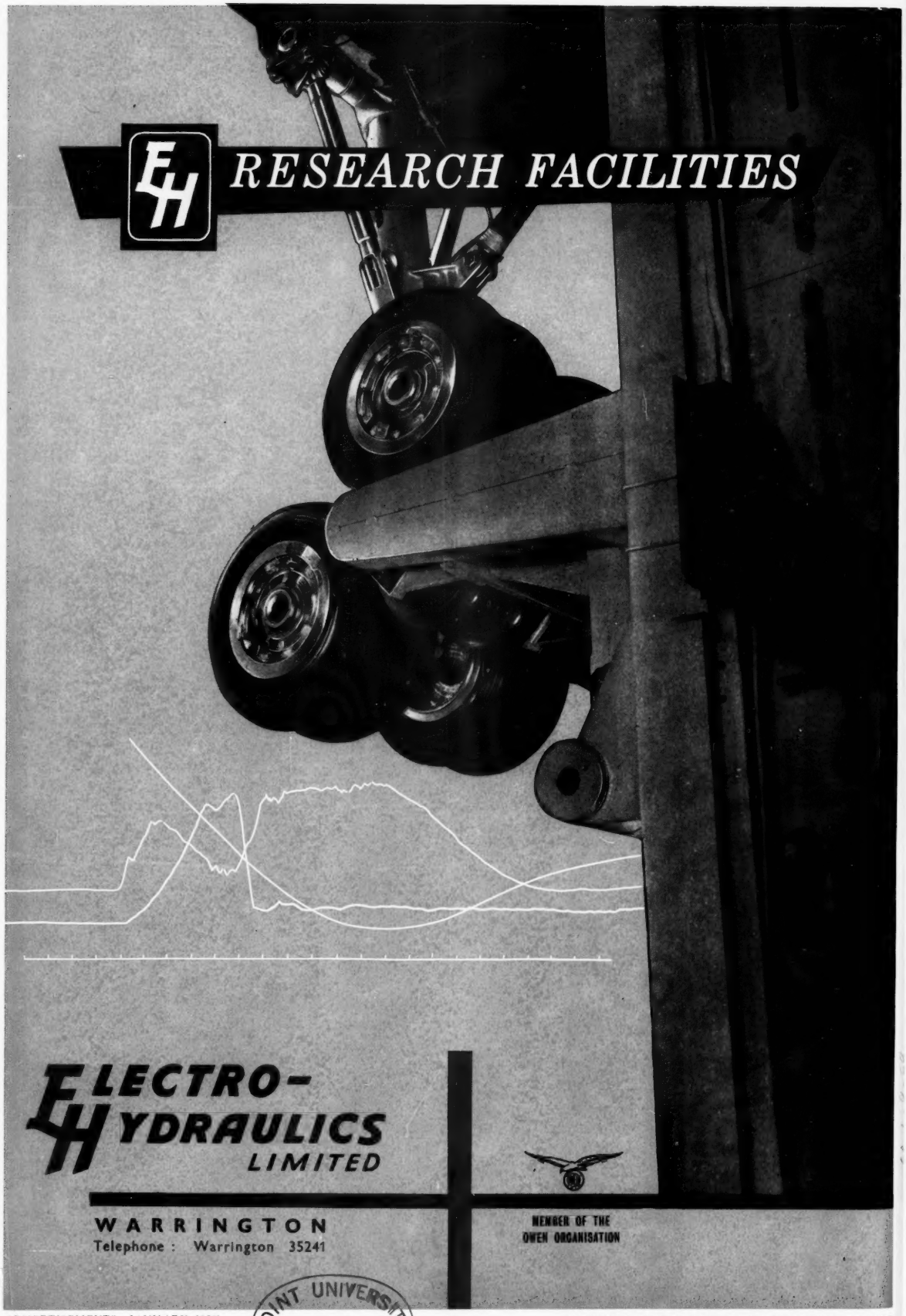

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Journal of
THE ROYAL AERONAUTICAL SOCIETY

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All communications for publication in, and advertisements for, the JOURNAL *should be addressed to the Editor,* JOAN BRADBROOKE, *A.R.Ae.S.; all communications on general matters affecting the Society should be addressed to the Secretary,* A. M. BALLANTYNE, *T.D., B.Sc., Ph.D., Hon.F.C.A.I., A.F.I.A.S., F.R.Ae.S.*

None of the papers or paragraphs must be taken as expressing the opinion of the Council of the Royal Aeronautical Society unless otherwise stated.

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The Royal Aeronautical Society

(Founded 1866)

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HER MAJESTY THE QUEEN

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The Journal of the Royal Aeronautical Society

WITH WHICH IS INCORPORATED THE INSTITUTION OF AERONAUTICAL ENGINEERS

From the President

December 1957

THE Society can be justly proud in the knowledge that much of the scientific and engineering work which made possible many of the military and civil aeronautical developments of the world has in fact been done in this country by men and women who are members of our Society.

The year 1958 may well be the starting point of a new era in Aviation which will bring with it great challenges and many difficulties. The formation of the Guided Flight Section during 1957 is evidence that our Society is prepared and determined to play its part in the future as it has in the past.

1957 has again been a good year marked by significant increases in membership. With your help, 1958 will be an even better year for the Society.

Best Wishes to you all.

George Edwards.

NOTICES

Presidential Address

Wednesday 26th February 1958

Sir George R. Edwards will deliver his Presidential Address on Wednesday 26th February 1958 at Church House, Westminster, London, S.W.1, at 6.0 p.m. The address will be followed by a Reception at 4 Hamilton Place, London, W.1, and full particulars of the arrangements and the Reception will be sent to all members.

Section Lecture Cancelled

It is regretted that the Section Lecture on Tuesday 4th February by Mr. G. Selwood on "Power Controls for Supersonic Aircraft" *has had to be cancelled.*

Guided Flight Section

Many members will have seen the large poster, issued in October, giving details of the Guided Flight Section of the Society, but for the benefit of those who have not seen it the following are the main particulars of the Section:—

The Section will be under the general direction of the Council, within the Charter and By-Laws of the Society, but will enjoy a large measure of autonomy in the conduct of its affairs. To inaugurate the Section a Steering Committee was formed which has now become the Section's first Committee. This Committee is as follows:—

Dr. G. W. H. Gardner, C.B., C.B.E., B.Sc., F.R.Ae.S. (Chairman), Director, Royal Aircraft Establishment.
Mr. H. H. Gardner, B.Sc., F.R.Ae.S., Director and Chief Engineer, Military Aircraft, Vickers-Armstrongs (Aircraft) Ltd.
Mr. M. J. Brennan, B.Sc., F.R.Ae.S., Chief Designer, Saunders-Roe Ltd.
Dr. R. R. Jamison, B.Sc., A.F.R.Ae.S., Engineer-in-Charge Ram-Jet Development, Bristol Aero-Engines Ltd.
Mr. H. C. Pritchard, B.A., F.R.Ae.S., Deputy General Manager, Elliott Bros. (London) Ltd.
Mr. G. G. Roberts, M.Sc., A.F.R.Ae.S., Technical Director, Smiths Aircraft Instruments Ltd.

The Committee will normally be elected by the members of the Section in a postal ballot. This election will be held in the Spring of 1958. Proposals for Membership of the Committee will be announced later.

The Section's first lecture was given on 21st November 1957 by Mr. J. Serby, C.B.E., B.A., F.R.Ae.S., Director-General of Guided Weapons, Ministry of Supply, on "Guided Weapons and Aircraft." This lecture was held at the Royal Society of Arts, Albemarle Street, and was presided over by the President, Sir George Edwards.

Mr. Serby's lecture will be published in an early issue of The Journal and it is intended that all the Section's Lectures and Notices will be published in The Journal.

The following is the Section's programme for the Spring 1958 Session which includes those main Lectures of the Society of direct interest to the Section.

Thursday 23rd January 1958

Guidance and Control. L. H. Bedford, C.B.E., M.A., B.Sc., Chief Engineer, English Electric Co. Ltd.
The Library, 4 Hamilton Place, at 7.0 p.m.

Tuesday 25th February 1958

Guided Flight Trials. R. W. Boswell, O.B.E., Deputy Controller, Controls and Instruments, Weapons Research Establishment, Salisbury, South Australia.
The Library, 4 Hamilton Place, at 7.0 p.m.

Thursday 20th March 1958

Problems in the Development of a Guided Missile. J. Clemow, M.A., Chief Engineer, Weapons, Vickers-Armstrongs (Aircraft) Ltd.
The Library, 4 Hamilton Place, at 7.0 p.m.

Thursday 10th April 1958

Guided Weapons. R. Cockburn, C.B., O.B.E., Ph.D., Controller of Guided Weapons and Electronics, Ministry of Supply.
The Institution of Mechanical Engineers, Birdcage Walk, at 6.0 p.m.

Thursday 15th May 1958

46th Wilbur Wright Memorial Lecture.
Automatic Flight. Dr. G. W. H. Gardner, C.B., C.B.E., B.Sc., F.R.Ae.S., Director, Royal Aircraft Establishment.
The Institution of Mechanical Engineers, Birdcage Walk, at 6.0 p.m.

It is the intention of the Committee to bring before the members of the Section, through lectures, discussions, and symposia, developments in guided missiles and automatic flight as well as in space travel.

Members of the Society interested in the Guided Flight Section are invited to register as members of the Section. Members of the Section must also be members of the main Society; those wishing to join should apply to the Secretary of the Society.

Nominations for Fellowship of the Society

The attention of Members is drawn to the following extract from the Society's By-Laws:—

"Fellows shall comprise every person who on the 30th day of January 1950 was on the Register as a Fellow of the Society; and every person thereafter awarded the honour of Fellowship.

"Elections to Fellowship will be made annually by the Council and will be announced at the Annual General Meeting of the Society. Nominations will be initiated by the members of the Council or by any four Fellows of the Society. It is the duty of the Council to see that the honour is awarded only to persons who have attained a considerable degree of technical eminence in the profession of aeronautics.

From this, it will be seen that any four Fellows may nominate a suitable candidate for Fellowship. Nominations for 1958 must be received by the Secretary **by 31st January 1958.**

Air Transport Course—Oxford

23rd March–3rd April. 8th April–12th April

The Third Air Transport Course will be held at Oriel College, Oxford, from the 23rd March to the 3rd April, with a break for the Easter week-end 4th, 5th, 6th and 7th April. The Course will re-assemble on Monday evening 7th April, then continue from the 8th April to the 12th. Full particulars may be obtained on application to the Secretary of the Society. The Course will follow the lines of previous courses with lectures on Operations, Air Law and Economics and there will again be a number of guest lecturers.

News of Members

J. W. Baumberg (*Graduate*), of Fairey Aviation Co. Ltd., Hayes, is now a Stressman with the Fairey Aviation Co. of Canada Ltd., in Nova Scotia.

Squadron Leader L. J. G. Bayliss (*Associate*), formerly of Fielding and Platt Ltd., has taken up an appointment as Technical Author with Dowty Fuel Systems Ltd., Cheltenham.

T. H. Blake (*Associate Fellow*), formerly at de Havilland Aircraft Co. Ltd., has been transferred to the post of Senior Inspecting Officer (Aircraft), A.I.D., M.O.S., London.

Squadron Leader H. W. Chiverton (*Associate*), formerly Senior Engineer Officer, R.A.F., Gutersloh, Germany, is now Technical Plans Staff Officer, Bomber Command H.Q.

DIARY

LONDON

15th January 1958

MAIN LECTURE AT LEICESTER BRANCH.—Aircraft Engines. A. A. Lombard. Lecture Theatre, Loughborough College. 6.45 p.m.

21st January

SECTION LECTURE.—Environments and Environmental Testing. P. J. Duncton and T. F. R. George. Library, 4 Hamilton Place, W.1. 7 p.m.

4th February

SECTION LECTURE.—Power Controls for Supersonic Aircraft. G. Selwood. Library, 4 Hamilton Place, W.1. 7 p.m. *This lecture has been cancelled*

13th February

MAIN LECTURE.—The Application of the Theory of Stability in Structural Design. H. L. Cox. Institution of Mechanical Engineers, Birdcage Walk, S.W.1. 6 p.m.

18th February

SECTION LECTURE.—Aeroelastic Problems Associated with High Speeds and High Temperature. E. G. Broadbent. Library, 4 Hamilton Place, W.1. 7 p.m.

20th February

MAIN LECTURE AT HALTON BRANCH. THE FIRST TRENCHARD MEMORIAL LECTURE.—Air Chief Marshal Sir Dermot Boyle. Branch Hut, R.A.F. Halton. 6.45 p.m.

26th February

PRESIDENTIAL ADDRESS. Sir George R. Edwards. Church House, Westminster, S.W.1. 6 p.m. Followed by reception at 4 Hamilton Place. Details to be announced.

GUIDED FLIGHT SECTION

23rd January

GUIDANCE AND CONTROL. L. H. Bedford. Library, 4 Hamilton Place, W.1. 7 p.m.

25th February

GUIDED FLIGHT TRIALS. R. W. Boswell. Library, 4 Hamilton Place, W.1. 7 p.m.

GRADUATES' AND STUDENTS' SECTION

28th January

DEVELOPMENTS IN GLIDER DESIGN. K. G. Wilkinson. Library, 4 Hamilton Place. 7.30 p.m.

BRANCHES

13th January

Glasgow.—Guided Weapons. J. J. Gait. Room 24, College of Science and Technology, Duke Street, Glasgow. 7.15 p.m.

Halton.—Impressions of Soviet Aviation. C. Jenkins. Branch Hut, R.A.F. Halton. 6.45 p.m.

Henlow.—Procedures of Test Flying at Supersonic Speeds. Wg. Cdr. R. P. Beamont. Building 62, R.A.F. Technical College, Henlow. 7.30 p.m.

14th January

Boscombe Down.—Aircraft Control. Prof. G. A. Whitfield. Lecture Hall, A. and A.E.E. 5.45 p.m.

Cambridge.—Medical Aspects of Modern Aviation. Wg. Cdr. Lewis. No. 1 Lecture Theatre, Cambridge University Engineering Laboratories. 8.15 p.m.

15th January

Brough.—Aviation's Prophets. Capt. J. L. Pritchard. Lecture Hall, Electricity Offices, Ferensway, Hull. 7.30 p.m.

Chester.—Nuclear Propulsion of Aircraft. R. E. Wigg. Lecture Theatre, Grosvenor Museum, Chester. 7.30 p.m.

Leicester.—Main Society Lecture. Aircraft Engines. A. A. Lombard. Lecture Theatre, Loughborough College. 6.45 p.m.

16th January

Christchurch.—The International Geophysical Year. Dr. R. d'E. Atkinson. Town Hall, Christchurch. 7.30 p.m.

Yeovil.—Production Techniques. P. Hyde. Park School, Park Road, Yeovil. 7.30 p.m.

18th January

Birmingham.—Airworthiness for the A.R.B. W. Tye. Engineering Centre, Birmingham, 7.30 p.m.

20th January

Halton.—Film Night. Branch Hut, R.A.F. Halton. 6.45 p.m.

21st January

Luton.—New Methods of Approach to Airworthiness. W. Tye. Napier Senior Staff Canteen, Luton Airport. 6.15 p.m.

22nd January

Hatfield.—Submarine Design Problems. E. A. Brokensha. de Havilland Restaurant, Hatfield. 6.15 p.m.

23rd January

Isle of Wight.—A Bush Pilot in the Far North. R. J. P. Barber. Clubhouse, Saunders-Roe Sports and Social Club, E. Cowes. 6.30 p.m.

27th January

Halton.—Film Night. Branch Hut, R.A.F. Halton. 6.45 p.m.

28th January

Bristol.—Aircraft Flight Systems. Capt. A. M. A. Majendie. Filton House. 6 p.m.

29th January

Preston.—Ram Jets. R. R. Jamison. Queen's Hotel, Lytham. 7.30 p.m.

Southampton.—Flight Testing Procedures at Supersonic Speeds. Wg. Cdr. R. P. Beamont. Institute of Education, University of Southampton. 7 p.m.

Weybridge.—Film Show. Apprentice Training School, Vickers-Armstrongs (Aircraft) Ltd. 6 p.m.

31st January

Chester.—Annual Dance. Quaintways Restaurant, Chester.

3rd February

Derby.—Annual General Meeting. Ten-minute Papers. Rolls-Royce Welfare Hall, Nightingale Road, Derby. 6.15 p.m.

Henlow.—Annual General Meeting. Building 62, R.A.F. Technical College, Henlow. 7.30 p.m.

4th February

Glasgow.—Pilcher Memorial Lecture. History of Scottish Aviation. Room 24, College of Science and Technology, Duke Street, Glasgow. 7.15 p.m.

Luton.—Development of the Armstrong Siddeley Viper. W. H. Lindsey. Napier Senior Staff Canteen, Luton Airport. 6.15 p.m.

5th February

Brough.—Supersonic Aircraft and Missiles. P. J. Duncton. Lecture Hall, Electricity Offices, Ferensway, Hull. 7.30 p.m.

London Airport.—Noise. Professor E. J. Richards. Apprentice Lecture Room, Fairey Aviation Co. Ltd., Hayes. 6 p.m.

Reading.—Ultra Sonic Testing. J. Crowther. Upper Canteen, Western Manufacturing (Reading) Ltd. 6 p.m.

6th February

Bristol.—Design of Civil Aircraft. D. P. Thorne. Filton House. 6 p.m.

Isle of Wight.—The Development of the Rotodyne and Gyrodyne. D. M. Davies. Clubhouse, Saunders-Roe Sports and Social Club, E. Cowes. 6.30 p.m.

Swindon.—Joint Lectures with Swindon Engineering Society. Gas Turbines as applied to Rail Traction. R. J. Harding, **and** Aircraft Gas Turbines. A. C. Lovesey. The College, Victoria Road, Swindon. 7 p.m.

Yeovil.—Voyage to Virginia. W. Courtenay. Park School, Park Road, Yeovil. 7.30 p.m.

11th February

Boscombe Down.—Film Evening. Lecture Hall, A. and A.E.E. 5.45 p.m.

12th February

Chester.—Joint Meeting with Stanlow Branch of Institute of Petroleum. Aircraft Fuelling Problems. E. L. Bass. Lecture Theatre, Grosvenor Museum, Chester. 7.30 p.m.

13th February

Cheltenham.—The Microscope in Engineering. A. E. Bingham. St. Mary's College, Cheltenham. 7.30 p.m.

Bristol.—Production of Civil Aircraft. By a member of Vickers-Armstrongs (Aircraft) Ltd. Filton House. 6 p.m.

17th February

Henlow.—N. E. Rowe Medals—Junior Members Lecture Competition. Building 62, R.A.F. Technical College. Henlow. 7.30 p.m.

19th February

Christchurch.—Aircraft Carriers. A. J. Simms. Kings Arms Hotel. 7.30 p.m.

Leicester.—The Activities of the P.E.R.A. Organisation. R. Tilsley. Lecture Theatre, Loughborough College. 6.45 p.m.

Southampton.—Branch Prize Papers. Institute of Education, University of Southampton. 7 p.m.

20th February

Bristol.—Sale of Civil Aircraft. By a member of Bristol Aircraft Ltd. Filton House. 6 p.m.

Halton. — MAIN SOCIETY LECTURE. First Trenchard Memorial Lecture. Air Chief Marshal Sir Dermot Boyle. Branch Hut, R.A.F. Halton. 6.45 p.m.

DIARY—Cont.

20th February

Isle of Wight.—Modern Methods of Air-Sea Rescue. Lt. Cdr. J. S. Sproule. Clubhouse, Saunders-Roe Sports and Social Club, E. Cowes. 6.30 p.m.

Weybridge.—Some Interesting Problems in Ship-Building Today. R. J. W. Rudkin. Apprentice Training School of Vickers-Armstrongs (Aircraft) Ltd., Weybridge. 6 p.m.

21st February

Birmingham.—Branch President's Night. F. F. Crocombe. Engineering Centre, Birmingham. 7.30 p.m.

26th February

Hatfield.—The Fairey Delta 2. R. L. Lickley and L. P. Twiss. de Havilland Restaurant, Hatfield. 6.15 p.m.

27th February

Bristol.—Servicing of Civil Aircraft. I. Gregory. Filton House. 6 p.m.

Cambridge.—Some Aspects of Aircraft Production. H. Povey. No. 1 Lecture Theatre, Cambridge University Engineering Laboratories. 8.15 p.m.

Yeovil—THIRD HENSON AND STRINGFELLOW MEMORIAL LECTURE. S. G. Hooker. Park School, Park Road, Yeovil. 7.30 p.m.

ELECTIONS

The following is a list of new members and transfers of membership of the Society:—

Associate Fellows

Beryl Edna Beadle (*from Graduate*)
Frank Bennison (*from Graduate*)
Richard Eric Burns
Brian Cape
Denys Sproston Challener
Ronald Wheeler Cross (*from Graduate*)
Geoffrey Bernard Dellora
John Stewart Drabble (*from Graduate*)
Joseph Brian Edwards (*from Graduate*)
John Hugo Ford
Ronald John Frost (*from Graduate*)
Maurice Alfred Gee (*from Student*)
Michal Jan Henryk Giedroyc (*from Graduate*)
George Calvin Grogan
Joseph Edward Hart (*from Graduate*)
David Roy Hooper
Charles George Buchanan McClure (*from Associate*)
William Makinson
Waclaw Markowski
Derek Graham Mather
Chandrakant Shridhar Naik
Michael O'Connor (*from Graduate*)
Thomas Robert Howard Parkes (*from Graduate*)
David Thornton Parry
Albert Ernest Peart
John Pickles (*from Graduate*)
Rhys Price Probert
David Henry Raynes
John Alexander Ritchie (*from Graduate*)
Michael Edmund Lock Spanyol (*from Graduate*)
John Harold Spaull
David Allan Spence
John William Hugh Thomas
Wilfred Roy Thomas
Stephen Frederick Wheatcroft
Frederick George Arthur Williams (*from Graduate*)
Bernard Ramsay Woods
Samuel Wroath

Associates

Clarence Herbert Baker
John Beattie
Arthur William Bennett
Walter John Bird
Leslie Broadhurst
Charles Arthur Philip Broomhead
Richard Allan Burberry
John Russell Burchall
Richard Anthony Buxton
James Reginald Calder
Clifford Barry Cox
William James Rixon Cross
Ralph Hunter Dargue
Deryck Aubrey Elliott
Peter Thomas Elliott
John Edward Garton
John Meredith Gray
Frank Arthur Harrington
Arthur Thomas Hills (*from Student*)
Noel Harry Hooke (*from Student*)
Ian Douglas Huntley (*from Student*)
Edward Charles Jackson
Charles Eric Johnson
Geoffrey Charles Kendell
Kenneth Charles Lucas
Roy John Williamson Marsh
Donald George Maskrey (*from Student*)
Kieran Thomas McKenzie (*from Student*)
Charles Edwin McLaurin
Maurice William Melville
Michael Nowell Morss (*from Student*)
Stephen Pearcy
Basil Thorpe Ratcliffe
William Kenneth Sharp
John Frederick Spurden
Ronald George Thomas
James Henry Woodward
Reuben Yorke

Graduates

Stanley Arthur Beasley
Michael Jeremy Bennison (*from Student*)
Frederick Nicholas Birkett (*from Student*)
John Martin Brant
Charles Mowat Christie
Trevor William Claxton (*from Student*)
Peter John Cutt
Brian John Edwards (*from Student*)
Richard Bryan Erb
Paul Anthony Foley
Peter Robin Freeman
Denis Arthur Frith
Anthony Neil Gobby
Gordon Hugh Kerr Goold (*from Student*)
Michael John Kingston
David Overington Lewis
Woon Lyloc
John Carlton Moore
Joan Napper
Gerald Ivor Pauley (*from Student*)
Gerald Andrew Prince
Bernard Reuben
Dennis George Roberts
Michael Graham Shaw (*from Student*)
Charles Myron Shure
Derek Murray Squires (*from Student*)
Owen James Stevenson (*from Student*)
Charles Philip Swift (*from Student*)
Ian Charles Thomas
Duvvuri Tirumalesa
John Stafford Toms
Frederick John Wild
Alan Keveral Wrigley

Students

Walter Geoffrey Benson
Balwant Singh Bhandari
Victor Frederick Bignell
David Brendon Thomas Birch
Edward Alfred Cant
John Robert Cockerill
Edward Howard Collier
Roy Collins
Thomas Allan Collinson
Robert James Cook
George Fredrick Crudgington
Peter Geoffrey de Bourcier
William Bruce Somerville Dobie
Anthony Robert Dudley
James Watson Failes
Brian David Freeman
John Gaughan
John Andrew Greenwood
David Henry Griffin
Andre Emile Gros-Balthazard
Gregory Quy Hall
Terry Alexander Peter Hamilton
Albrecht Hartmann
Robert Lewis Hyde
Ian Vaugton James
David Leslie Key
Moiz-Uddin Khan
John Kinsey
William James Lavington
Douglas William Lean
Graham Houston MacDonald
Alan Milner
Ronald Thomas Mold
Colin Charles Morath
Lance Arthur Osborne
Alastair Robert Paterson
John Derek Preston
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TRANSONIC WIND TUNNEL TESTING TECHNIQUES

An All-Day Section Lecture and Discussion on "Transonic Wind Tunnel Testing Techniques" was held by the Society on 20th December 1956 at the Institution of Mechanical Engineers, 1 Birdcage Walk, London S.W.1, and was presided over by Major G. P. Bulman, C.B.E., B.Sc., F.R.Ae.S., Director of Construction and of Research Facilities, Ministry of Supply. Introductory papers were given by Mr. H. F. Vessey, B.Sc., F.R.Ae.S., Mr. J. A. Kirk, A.F.I.A.S., A.F.R.Ae.S., Mr. R. Hills, B.A., A.F.R.Ae.S., and Mr. F. E. Roe, B.Sc.(Eng.), D.I.C., A.C.G.I., G.R.Ae.S. There was a Discussion at the end of the morning session and at the end of the day. Mr. J. A. Dunsby, M.Sc.(Eng.), D.I.C., A.M.I.Mech.E., T.M.I.A.S., A.F.R.Ae.S., acted as rapporteur for the Discussions.

Historical and General Introduction

by

H. F. VESSEY, B.Sc., F.R.Ae.S.

(Royal Aircraft Establishment, Farnborough)

THIS IS the first meeting at which it has been possible to talk openly about transonic tunnels with ventilated working sections. Many already knew that a large number of tunnels, particularly in the U.S.A. and this country are being built or modified with slotted or perforated working sections. Up to quite recently, however, security considerations have shrouded this work but it is now possible to draw the curtain.

There are four things that ventilating the working section of a closed jet tunnel can do: —

1. Reduce (or eliminate) tunnel corrections at subsonic speeds, or alternatively allow larger models to be used for the same corrections.
2. Prevent choking at high subsonic speeds.
3. Establish a uniform tunnel flow from a Mach number of, say, 0·8 to 1·4.
4. Reduce or eliminate shock reflections at low supersonic Mach numbers.

The first possibility has been recognised for many years, in fact since the days of Glauert when it was noted that the corrections for a closed jet and an open jet tunnel were of opposite signs and therefore, that it might be possible to design a compromise partly open, partly closed, tunnel with greatly reduced corrections. Some work was done in various countries, examples of which are given in Refs. 1, 2 and 3, written between 1936 and 1941. In 1940 Sir Geoffrey Taylor suggested that blockage might be reduced and choking delayed in a high speed tunnel by ventilation, but unfortunately this proposal was not taken up.

The impetus for this work in the transonic field came from America and appears to originate from the tests of Wright and Ward[4] at N.A.C.A. described in October 1948 in what was then a classified document. From this time onwards more and more interest was taken in ventilated tunnels for transonic operation. Stack spoke in very guarded terms of the American work at the Joint R.Ae.S.-I.A.S. Conference at Brighton in 1951, and the curtain was further drawn at the I.A.S. Symposium "Transonic Testing Techniques" in 1954. The papers for this symposium[5] give a clear picture of the considerable activity on this subject at that time, although some of the details are lacking.

In this country some unpublished R.A.E. work was done as a preliminary investigation for a possible High Speed Tunnel conversion and Holder[6] at N.P.L. made an investigation of slotted and perforated walls on two sides of a 7½ in. × 3 in. induced flow tunnel. English Electric were first in the field with an operational tunnel, for their 12 in. × 12 in. was running in 1950. The larger tunnels followed more slowly, the Bedford 3 ft. × 3 ft. tunnel had an insert with two slotted walls operating in 1952 and has now converted to a four wall slotted configuration. The R.A.E. 10 ft. × 7 ft. high speed tunnel was re-opened by the Minister in April 1956 after conversion to an 8 ft. × 6 ft. transonic tunnel and in May 1956 the Duke of Edinburgh opened the Aircraft Research Association 9 ft. × 8 ft. transonic tunnel. The N.P.L. now has some five slotted tunnels of moderate size for specialised research and a number of aircraft firms are operating tunnels of from 1 ft. to 2 ft. square.

Early Tunnels

Early tunnels, and probably the majority of existing tunnels, use diffuser suction to establish flow through the ventilated walls. The more general case of auxiliary suction by means of a pump will, however, be described first. A diagrammatic sketch is shown in Fig. 1. The working section walls, which are slotted or perforated, are surrounded by a plenum chamber to which suction is applied to hold the pressure in it at approximately the desired working section static pressure. For supersonic operation the beginning of the working section is designed to provide a throat so that sonic speed is first reached there. The flow is then expanded supersonically by suction through the vented walls and with the appropriate suction and the correct pressure ratio from the main drive the Mach number distribution should be as shown, with the flow accelerating steadily to the desired Mach number, remaining constant over the model test section and terminating in a shock at the end of the working section. (This diagram can also be regarded as an inverse plot of the static pressure along the working section.) The plenum chamber pressure will closely approximate to the static pressure in the fully developed flow and there will be an outflow from the early portion of the vented walls. In the region of fully established flow there should be little flow through the walls. If, however, for example, there is a local drop in M (and hence rise in static pressure) outflow will take place and will tend to compensate. For supersonic flow where expansion causes an increase in Mach number the system is, therefore, highly stable. In the subsonic case the primary purpose of suction is to remove the boundary layer and so prevent choking but, unlike the supersonic case, the system is not stable in establishing uniform flow for a local drop in speed will cause an outflow which will cause a further drop in speed. Suction quantities are small however, and little difficulty is generally experienced due to this cause. Suction quantities can be calculated to a first approximation by assuming simple isentropic expansion from $M=1$ at the throat to the required Mach number in the working section. This, however, neglects the suction required to unchoke the tunnel at subsonic speeds before sonic speed is reached in the working section. By allowing for this quantity which can be calculated (again to a first approximation) by consideration of the amount of boundary layer to be removed, a better approximation is obtained.

Early tunnels, and many existing ones, use automatic suction obtained by venting the plenum chamber to a low pressure region of the tunnel, generally formed by modifying the diffuser. The slots or perforations may be continued into the diffuser as in Holder's experiments, but a more flexible scheme is that devised by Boeing's where the expansion is obtained by a step at the end of the working section and in which the amount of flow is regulated by flaps, as will be described by Mr. Kirk. An interesting variant of this scheme has been designed by Dr. Spence for the 2 ft. × 1½ ft. transonic tunnel shortly to run as an auxiliary to the R.A.E. 8 ft. × 6 ft. transonic

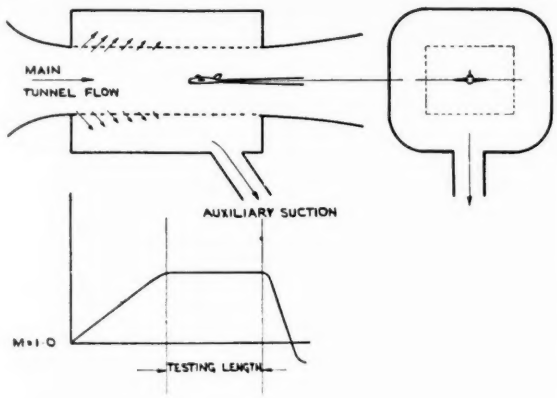

FIGURE 1. Transonic tunnel employing auxiliary suction.

tunnel. Following the working section a supersonic effuser accelerates the flow to a Mach number of about 2. At this point the air drawn through the slots enters by adjustable doors. This gives greater suction than other "automatic suction" schemes and allows a higher Mach number in the working section. It is, however, only possible because of the high pressure ratio available for this tunnel.

The choice of the actual arrangement used in any particular tunnel will be discussed by others but in general:—

1. Auxiliary suction by means of a pump is simple and leads to independent operation of controls. It requires rather less total power than diffuser suction but does require extra (and quite expensive) plant.
2. Diffuser suction leads to economy in installation but requires rather more total power than independent suction due to the inevitable "ejector" losses. It requires rather a high main drive pressure ratio and the entry of "tired" air into the diffuser will lower its efficiency and may cause breakaway if the design is not conservative.

As previously described, flow through the walls is used to provide the expansion required to increase the Mach number from unity at the beginning of the working section to the required value in the model test section. This can be avoided by using the appropriate supersonic nozzle at the beginning of the working section. N.A.C.A., Ames, are believed to be the first to use this system and Mr. R. Hills will be describing its use in the A.R.A. Tunnel. Briefly the scheme consists of a flexible supersonic nozzle the shape of which is altered to generate the required Mach number. The extreme accuracy in nozzle shape required for supersonic tunnels is not necessary in this case as the stabilising effect of the suction flow will improve velocity distribution and shock waves from nozzle defects will be damped out by the two or three reflections required before reaching the model position (provided the working section is long enough). By controlling Mach number independently of suction flow, the supersonic

nozzle appears to allow a better compromise on flow distribution, shock cancellation and blockage. It does, however, require a longer tunnel and a comparatively expensive installation to operate the flexible nozzle.

I have referred to the four things vented tunnels can do: —

1. Reduce tunnel corrections.
2. Prevent choking.
3. Establish uniform transonic flow.
4. Reduce shock reflections.

Tunnel corrections have been exhaustively discussed in current literature and the position is well summarised by Maeder and Wood[7]. Briefly, most of the corrections for the various types of interference can theoretically be eliminated by the choice of the correct open/closed area in the working section, or can be calculated by the use of approximations. Choking is avoided by almost any of the configurations which give reasonable transonic velocity distribution and in practice most of the effort is devoted to 3 and 4. The open area is chosen by theory, model tests and past experience to minimise corrections and then by experiment, aided by theory, the flow distribution is improved to the standard required. Typical values are 10 per cent open area for slots and 20 per cent for perforations.

The problems of flow establishment are best shown by typical curves such as those of Figs. 2 and 3, which are taken from practice. The principle of the ventilated tunnel is to persuade the main stream flow to expand gradually and to form in fact its own supersonic nozzle. The porosity must, therefore, be graded and for this reason slots are tapered and perforated walls have gradually increasing open area. Fig. 2 shows the effect of too sudden an increase in open area, over-expansion at the beginning of the slots causes a jump in Mach number followed by a depression in a series of highly damped waves. This effect is avoided by establishing the correct taper and the result should be as in Fig. 3.

At the rear of the testing length it is possible for the speed to drop off too rapidly, as at the right hand end of Fig. 2. This can happen at subsonic speeds due to the pressure field from the support rig behind the model, or at high subsonic speeds due to too low a pressure ratio from the main drive. At supersonic speeds the same effect can be given by the bow shock from the support rig. The cure is obvious but not always easy to apply.

The curves shown were obtained by a centre line static survey in the R.A.E. 8 ft. × 6 ft. transonic tunnel. The process of cut and try can be speeded up by tracing the disturbances back to the wall by putting in artificial disturbances by means of a step on the wall and observing where the effect shows up on the central static (they are not in fact transmitted at exactly the Mach angle). An over-expansion can then be cured by reducing the slot width or, by inserting a perforated plate behind the slot in which case the effective blockage can be varied over quite wide limits by altering the angle of the perforated plate to the air flow.

Another powerful control on velocity distribution along the working section is provided by wall divergence which, preferably, should be adjustable. At subsonic speeds without suction a small divergence is required to compensate for boundary layer growth. At transonic speeds a small convergence thins the boundary layer and reduces shock reflection, while at supersonic speeds convergence appears to have a stabilising effect on flow generation. Convergence does, however, increase the power required and if the most is to be obtained from a tunnel it will be necessary to diverge the walls to reach the maximum Mach number and to accept some deterioration in distribution. For this reason the adjustable walls in the R.A.E. 8 ft. × 6 ft. tunnel are to be motorised.

I have left discussion of shock cancellation to the last as this is probably the most controversial aspect of transonic tunnel design—it certainly leads to fierce arguments between the advocates of slots and perforated plates.

The shock wave from, say, the nose of a model is reflected by a solid wall and at low supersonic speeds will strike the model again. This results in a change of pressure distribution and of the forces and moments. The mechanics of shock reflection and cancellation are shown in Fig. 4. The main cause of the reflection is the pressure rise on the wall due to the deflected flow behind the shock wave. If the porosity of the wall is such that the flow through it produces a pressure drop which is equal to the pressure rise through the main shock, there will be no reflection while, if it is greater, there will be partial reflection in the form of an expansion wave. (Expansion waves will be reflected in the inverse manner.) In a slotted wall tunnel shock waves will be reflected as shocks from the slats and as expansions from the open area, reflection is at a slightly different angle in each case and it will, therefore, be seen that unless the slots are very close there is little prospect of shock cancellation, although there may be some attenuation. With perforated walls in which the holes are

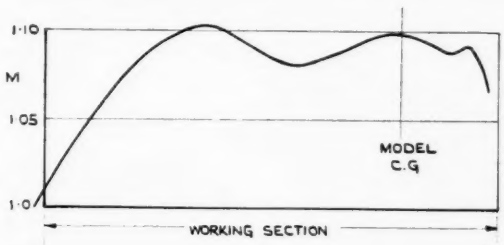

FIGURE 2.

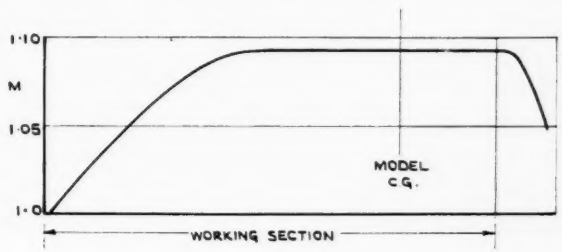

FIGURE 3.

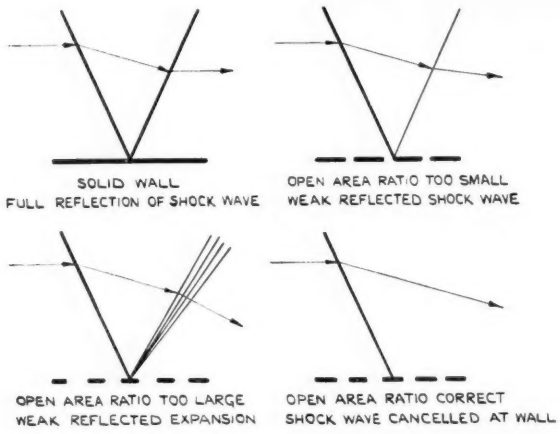
SOLID WALL
FULL REFLECTION OF SHOCK WAVE
OPEN AREA RATIO TOO SMALL
WEAK REFLECTED SHOCK WAVE
OPEN AREA RATIO TOO LARGE
WEAK REFLECTED EXPANSION
OPEN AREA RATIO CORRECT
SHOCK WAVE CANCELLED AT WALL

FIGURE 4. Shock wave reflection from perforated walls.

small in relation to the tunnel cross section, conditions are improved but as the theoretical open/closed area varies with Mach number and shock strength, the answer must, at best, be a compromise although it is claimed that acceptable results can be obtained over the working range by fixed wall porosity. It should be noted, however, that it is necessary to remove the whole of the boundary layer ahead of the shock or a complicated shock-boundary layer interaction causes shock reflection. Up to the present time, we in this country, have had little operating experience with perforated walls.

The choice between slots and perforations is a difficult one to make. Slots are claimed to be less difficult to install in an existing tunnel and to adjust for an acceptable velocity distribution. Perforations on the other hand promise some measure of shock attenuation but put up a pattern of a shock wave and an expansion wave from each hole, which may extend into the model area. Goethert claims that perforated walls require less power than slotted but bases this on a comparison of almost equal open areas. For practical operation slots require some half the open area of perforations and the powers then are much more equivalent.

The R.A.E. tunnels are slotted while A.R.A.'s tunnel, to be described by Mr. Hills, is perforated. When it can be arranged comparative tests in the four large British transonic tunnels will provide extremely valuable data. These have been planned and will take place as soon as possible.

Conversion of the R.A.E. Tunnel

Other papers will deal with the design of two types of new transonic tunnels and I should like to take this opportunity to say a few words on the conversion of the R.A.E. 10 ft. × 7 ft. "High Speed Tunnel," as the modification of an existing tunnel involves different

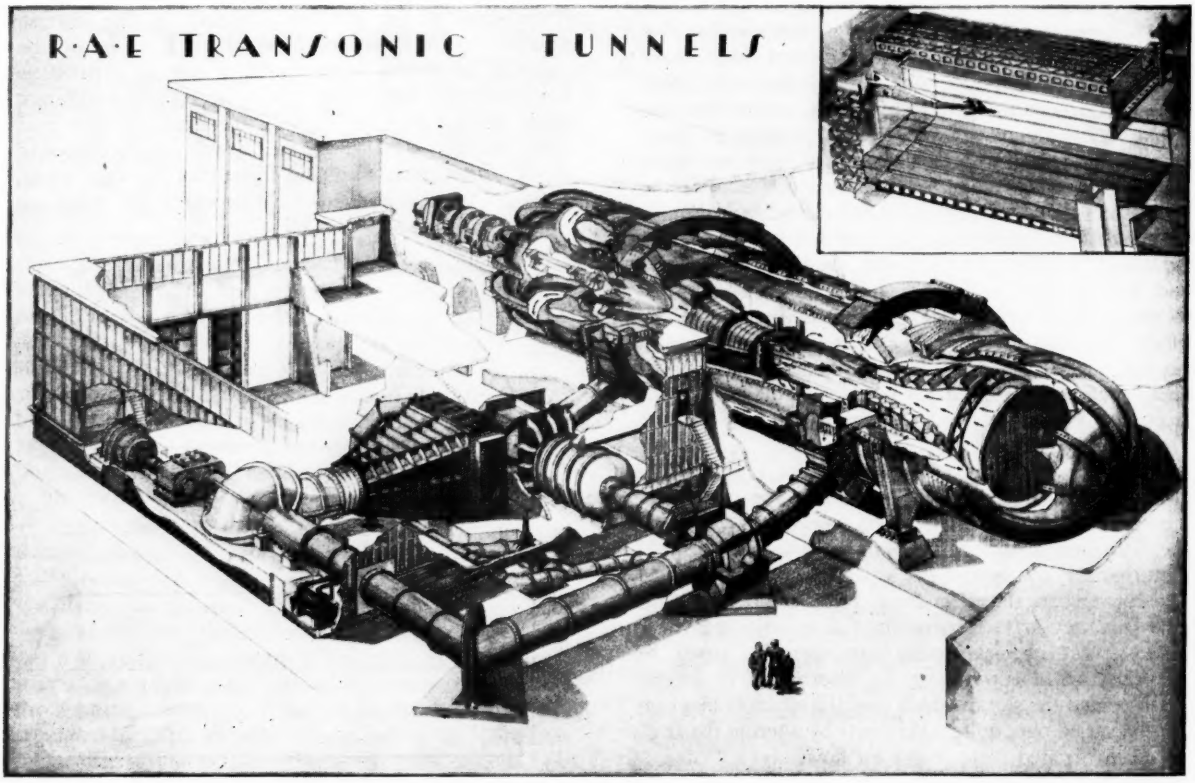

FIGURE 5.

problems to *ab initio* design. The tunnel started operation in 1942 and did most valuable work during the War and in the subsequent years. In 1951, Mr. Sterne started design work on its conversion to transonic operations. Modifications started in October 1954 and the tunnel is now in full operation.

The main drive consists of a single stage fan with a pressure ratio of the order of 1·1. Diffuser suction was, therefore, not possible and the choice of independent suction was made. Other factors influencing this decision were the desire to add power without increasing the fan loading and uncertainty on the behaviour of the diffuser and return circuit. The new arrangement is shown in Fig. 5. The old working section was cut out and replaced by a slotted one. What was previously the dead space is now used as a plenum chamber and the air is drawn out by ducts passing through the return circuit and connecting with an annular ring surrounding the outer shell. Ducts then lead to the auxiliary compressor room which was built on to the side of the old tunnel room. The compressor, driven by an 8,000 h.p. slip induction motor is rated at 4,000 cu. ft./sec. at an inlet pressure of 0·75 atmospheres absolute and a compression ratio of 2·7. After the compressor the air passes through a cooler and another annular ring to the tunnel return circuit which it enters with a small excess velocity.

The conversion was designed to give a Mach number of 1·2 (walls parallel) at a stagnation pressure of $\frac{3}{4}$ atmosphere and the working section was, therefore, reduced from 10 ft. × 7 ft. to 8 ft. × 6 ft. and the fan power increased from 4,000 h.p. to 12,000 h.p. This was done by coupling an 8,000 h.p. a.c. slip induction motor to the existing shaft in series with the two existing 2,000 h.p. d.c. motors. The two d.c. motors are used for automatic speed control and a load sharing electronic control transfers load from the d.c. motors to the new a.c. one when they take about 3,000 h.p.

Control of auxiliary suction is firstly by controlling the speed of the motor and secondly by by-passing a proportion of the air delivered by the compressor. The latter is the more sensitive control and is used for fine adjustment of the Mach number in the working section. While this adjustment is being made the motor speed is held constant by automatic speed control.

The working section is slotted on all four walls and the roof and floor are adjustable to give a maximum of $\pm\frac{1}{2}^\circ$ of convergence or divergence on each.

The model is observed by closed circuit television. Two cameras are used in the dead space looking through the slots and feeding two monitors in the control room. Normally one camera will be mounted on the side wall and the other on the roof to give side view and plan respectively, but the system is extremely flexible and the cameras can be used even in the return circuit if required. Tuft and evaporative flow visualisation can be clearly seen on the monitors and experience, so far, has been most encouraging.

The possibility of pressurising the tunnel for high Reynolds number tests has been retained and at a low Mach number $3\frac{1}{2}$ atmospheres (absolute) will give a

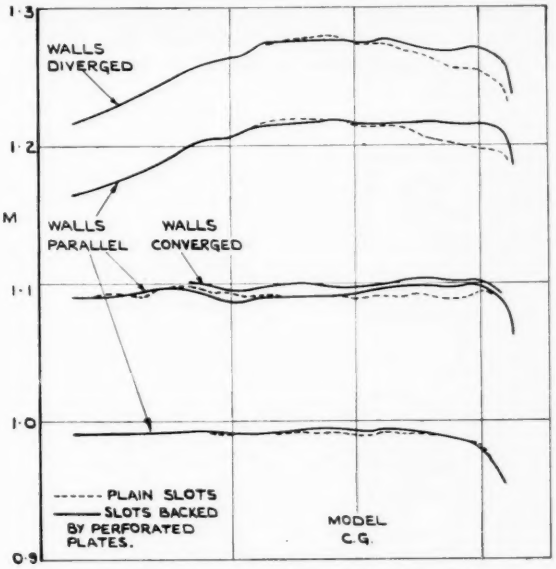

FIGURE 6. Typical centre line distributions. R.A.E. 8 ft. × 6 ft. transonic tunnel.

Reynolds number of 9×10^6 per foot. At higher speeds power requirements dictate a reduction of pressure and transonic runs will be made at a maximum of 0·8 atmospheres giving a Reynolds number of $3\frac{1}{2} \times 10^6$ per foot. Model strength may, however, dictate a lower stagnation pressure and at the extreme the pumps can hold the tunnel down to one-tenth atmosphere.

At the time of writing, preliminary calibration tests have been completed and we now have a mixed programme of model testing, flow improvement and final calibration. Results so far are very encouraging and some typical distributions are shown in Fig. 6. In the empty tunnel the maximum Mach number with walls diverged is 1·27 with a maximum error in the model region of $\pm 0{\cdot}007$. Up to $M = 1{\cdot}24$ the maximum error is $\pm 0{\cdot}004$. The maximum Mach number with walls parallel is 1·21. Pitch error is $\pm 0{\cdot}1^\circ$. We expect to be able to test a typical model at incidence at $M = 1{\cdot}2$.

Our planned programme of constraint models should indicate the size of model that can be tested. For the time being we are designing models that are required to be tested transonically to be less than 30 in. long so that the bow shock will be clear of the models by about $M = 1{\cdot}15$. Model blockage is being kept below 0·4 per cent.

Finally, it is worth mentioning that the auxiliary suction plant is also used to drive a small tunnel. A 2 ft. × $1\frac{1}{2}$ ft. slotted tunnel has been inserted between the main suction and delivery ducts and by suitable valves can be brought quickly into operation when suction is not required for the 8 ft. × 6 ft. tunnel. "Automatic" suction is used for flow expansion and takes the form of the supersonic effusor already described. Apart from this the working section is a model of the large tunnel and will be used in the first case for further development of the 8 ft. × 6 ft. tunnel. The slats for the side walls are made in glass so that

schlieren observations will be possible. The tunnel will be in operation shortly and is expected to run up to $M=1{\cdot}4$ at a stagnation pressure of $\frac{3}{4}$ atmospheres absolute.

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Design and Operational Problems of the Transonic Jet-Driven Wind Tunnel

by

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(de Havilland Aircraft Co. Ltd.)

Introduction

In this short paper on the transonic jet-driven tunnel, I will place the emphasis on the word transonic and deal only with the jet drive as it affects the design or operation of the transonic working section. I also would like to say that, although a fair amount of the information discussed afterwards will refer to the D.H. transonic tunnel, the problems are more or less general to the jet-driven tunnel and some will even be applicable to electrically-driven transonic tunnels using diffuser suction.

A fair amount of knowledge is now available on transonic working sections, but when the D.H. team started late in 1952 to look into the design of a suitable tunnel, very little information existed from British sources. Consequently, it was decided that we should seek the information required to complete our design from the U.S.A., and I would like to record here the generous help we received from all concerned during our short visit in April 1953 to the various firms and Government Establishments then operating, or designing, transonic tunnels.

Most people are probably familiar with the layout of the jet-driven wind tunnels as they have been the subject of previous lectures and have been described fairly fully in the technical press. Briefly, the jet engines are used as induction pumps to provide the necessary suction to drive the air; by judicious selection of the number of engines and of the mixing area ratio, a range of air speeds can be obtained in the working section. Once this range of speed has been selected, the working section area is more or less fixed (Fig. 1).

Because the power unit of the jet-driven tunnel is concentrated in one place, it has proved more convenient to use this power to generate the transonic flow in the working section instead of using auxiliary power. This immediately leads us to use the diffuser suction to generate the transonic flow, and the development seen in the U.S.A. during our visit suggested that the slotted working section designed by the N.A.C.A. Langley team was the most suitable, although it was clear, even at that time, that this slotted working section was not the complete answer to the problem.

Design Problems of the Transonic Working Section

As has already been mentioned, some of the requirements of the transonic working section are: —

(*i*) That it should provide continuous air stream from high subsonic speeds through $M=1$, without choking the tunnel.

(*ii*) That it should generate low supersonic flow.

(*iii*) That it should eliminate, or greatly reduce, the subsonic blockage interference.

(*iv*) That it should eliminate, or reduce, the constraint interference on lift.

(*v*) That it should eliminate or reduce the boundary interference, *i.e.* it should eliminate shock reflection from the walls.

(*vi*) That it should require the minimum amount of power to generate the required Mach number.

When we designed our tunnel, it was clear from American results that the slotted working section dealt very effectively with (*i*) and (*ii*). It was also clear that the subsonic blockage, together with the constraint interference, was greatly reduced. Item (*v*), *i.e.* elimination of shock reflection, did not appear too promising. However, by judicious design it was possible that the

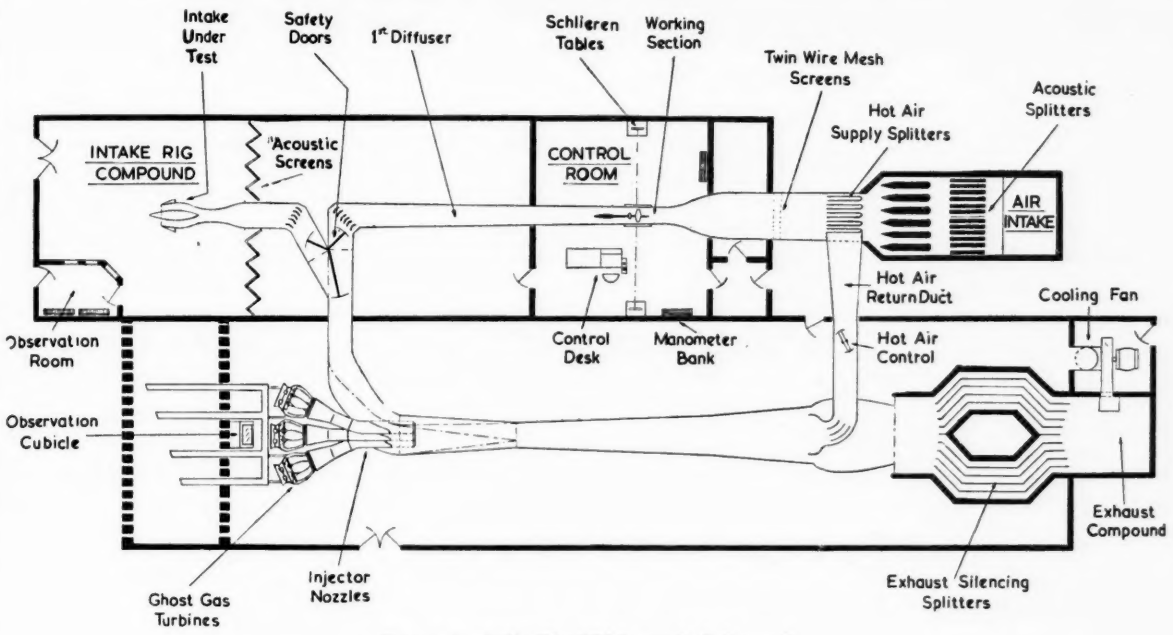

FIGURE 1. de Havilland high speed wind tunnel.

slotted working section could be made to run fairly economically.

We had then sufficient information to proceed with the design of our working section. The cross-sectional area had been fixed at 4 square feet by the mass flow available because it was required that the tunnel should also run at supersonic speeds up to $M = 1{\cdot}6$ with the use of solid liners (Fig. 2). It was also decided to have slots only on the top and bottom surfaces, and not on the sides. This requirement was mainly selected because it was suspected that the transonic section would not be very effective in eliminating shock wave reflections and it was therefore imperative that the model should be viewed constantly with schlieren, in an effort to decide when shock wave reflections were affecting the results. Hence, the two vertical walls had to incorporate 20 in. diameter windows and could not therefore accommodate slots.

It was also thought that a substantial flow passage had to be provided under the swept surface of the transonic liners to allow for flow expansion and a 12 in. space was provided above and below the ceiling and floor of the working section.

The next consideration was the slot shape. Experiments had shown that the flow had to be generated slowly and there was a great danger of creating over-expansions giving a local Mach number much greater than that desired. This would promote shock waves ahead of the working section, and the reflection of these waves from the tunnel boundary, travelling downstream to the model location, would give a non-uniform flow distribution near the model.

To counteract this effect a tapered slot shape was selected with a very small entry angle to avoid over-expansion. Also, because it was suspected that some variations to the shape of the slots would have to be made during the initial calibration, the working section was made of a series of *I* beams, bolted together. A thin metal sheet skin, which incorporated the slots, was placed on the top of each *I* beam to form the wind swept surfaces. If alterations were required to the slots it would then be quite easy to dismantle the liners and alter the slots. In fact, this did not prove necessary, as will be shown later.

To control the amount of flow out of the slots the convenient method of re-entry door, or Boeing flap as it is also called after the Boeing Airplane Company who first designed it, was used. This consists of a flap hinged at its rear so that by varying its angle the amount of flow out of the slots can be controlled. In our case we incorporated this flap with the centre liner required with the supersonic liners, with the idea that if a second throat were necessary, for either subsonic or high supersonic operation, it could be made, using this flap hinge.

The last design problem to be settled was the attachment of the liners, because in our tunnel, together with the transient liner, we also had to make provision for the $M = 1{\cdot}4$ and $M = 1{\cdot}6$ solid liners. It was decided to support the liners by two screw jacks, one near each end of the liner, the first one pulling the leading edge on the front knuckle so that a fairly good joint, with a very small step, would be made there; the second or rear jack was to allow the liner to tilt about the forward knuckle so that a certain amount of convergence or divergence could be incorporated to the liner setting. This provision was made for three purposes: —

(*i*) To provide a rapid method of altering the angular setting of the liners to allow for the boundary layer growth on its surface and to

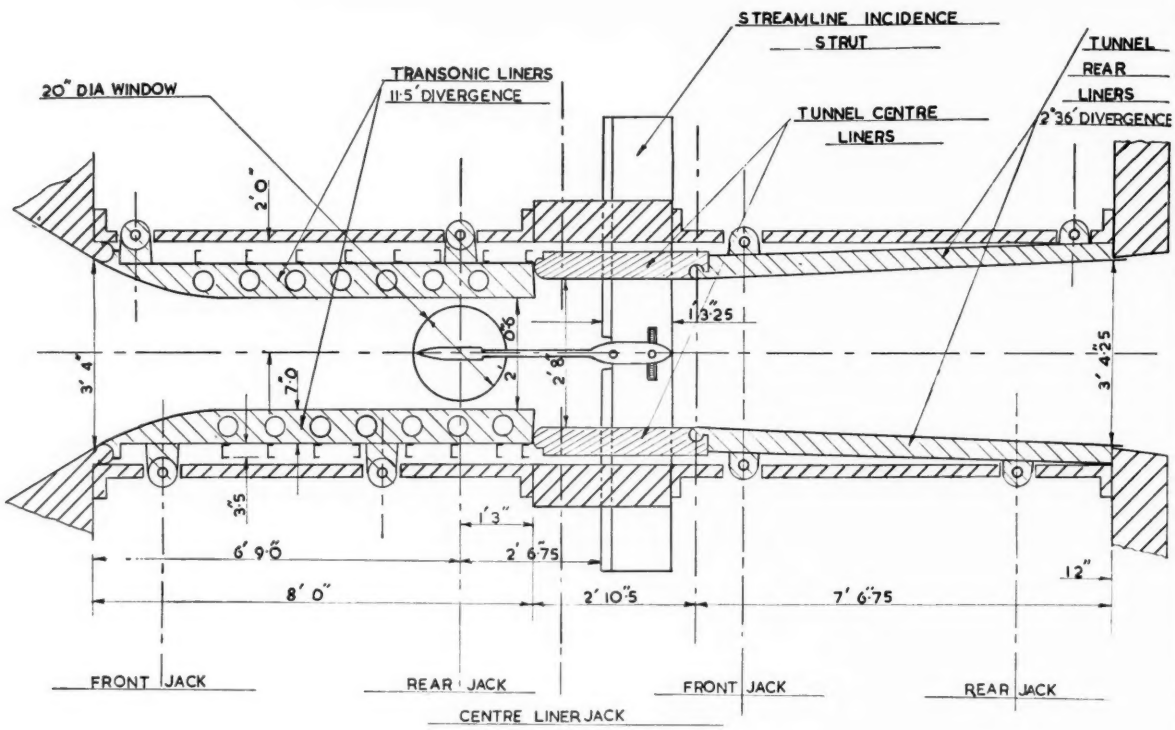

FIGURE 2.

eliminate the use of shims used in many supersonic tunnels to set the liners at the required divergence.

(*ii*) To experiment later with convergence, as it then appeared a possible method of reducing shock wave reflection.

(*iii*) To see if testing could be made on either side of the design Mach number of the solid supersonic liners by converging or diverging them. Fig. 3 shows the general layout of the control room and the working section with one side door removed.

Operational Problems of the Transonic Working Section

The main problems encountered during the calibration and operation of the tunnel are: —

(*i*) Temperature problems and provision of condensation free flow.

(*ii*) Provision of even Mach number distribution.

(*iii*) Measurement of pitch in the air stream of the working section.

(*i*) TEMPERATURE PROBLEMS

It is a well known fact that as the speed of the air is increased in a subsonic nozzle, the static temperature decreases and the moisture normally in suspension in the air stream will condense. The usual way of dealing with this problem, in conventional return circuit tunnels, is to fit a chemical dryer in the air stream which will remove most of the water content of the air.

In non-return circuit tunnels, because of the mass of air passing through the working section, this solution is not so attractive: the size of the drier which would be required for a continuous running tunnel being uneconomical. In our case, as we had a convenient source of heat available, *i.e.* the jet engines, it was thought more economical to raise the temperature of the air stream by ducting a proportion of the exhaust gases-air mixture just before they were released into the atmosphere and to recirculate it with the cold air ahead of the working section. A fair amount of minor modification had to be made before a reasonably even temperature distribution could be obtained in the working section.

Unfortunately, to achieve a condensation free flow in the working section, we have to run the tunnel at about 80°C for subsonic operation, 100°C to 120°C for transonic operation and up to 160°C for supersonic operation at $M = 1{\cdot}6$. Quite a few subsidiary problems are caused by these high temperatures and they had to be solved. They included the following:

(*a*) *Temperature shock on the schlieren windows*
These windows are made of $1\frac{1}{2}$ in. thick annealed glass of special quality and the makers recommend to keep the temperature shock below 30°C. Obviously this is not practical because of the time it would take to warm up the tunnel. Therefore, the glass has to be warmed much quicker, but in order to keep an eye on

the temperature rise a small thermocouple has been fitted in the bevel of the glass, on the wind swept surface.

Up to now, we have only had one window failure and as this happened not at the beginning of a run, but after about 20 minutes operation, I cannot believe that this failure was due to temperature stresses inside the glass, especially as these windows had sustained a much greater temperature rise, inadvertently, earlier in their life. My personal opinion is that the failure was due to some type of fatigue problem associated either with cyclic temperature rise or even cyclic loading, but I am told that this cannot be so because glass does not suffer from fatigue.

(*b*) All the internal strain gauge balances we manufacture have to be designed with this operating temperature in mind: very real difficulties are encountered and each strain gauge bridge has to be compensated for temperature drift. The design itself must make allowances for differential heating along the balance, as it is easy with a bad design to obtain a greater signal from the thermal expansion of the balance, than from the loading stresses experienced. The electrical wiring of the balance must also be made with wire which can sustain high operating temperatures.

(*c*) All parts of the models must be attached by screws, as we cannot use soft solder or any of the Araldite type of materials to joint structural members. These materials become sufficiently plastic at our operating temperature, and can be dislodged by the air forces experienced by the model.

It is also a requirement that the various parts of the model must be made of the same material to eliminate discontinuities that would arise because of the differential expansion of the parts.

(*ii*) MACH NUMBER DISTRIBUTION

When the calibration of the air stream in the working section was started, a serious over-expansion was experienced at the beginning of the slot, although the slot entry angle had been kept extremely small. However, by this time some information was available from the Royal Aircraft Establishment, Bedford, which suggested that such an over-expansion could be cured by fitting perforated plates in the slots, below the actual liner surfaces. These perforated plates worked very well and no other alterations were made to the slot shape.

Various tests were made to find the effect of the re-entry flap, and of the convergence and divergence of the liners on the flow distribution in the working section. It was found that, provided a second throat was not created at the rear by the hinge of the re-entry flap, the position of this flap had only a very minor effect. With the main liners it was found that convergence and divergence did not improve the flow, except at the highest Mach number obtainable with the transonic working section, *i.e.* at $M=1{\cdot}25$ and $M=1{\cdot}3$. It was therefore decided that for operations below $M=1{\cdot}25$ the normal setting of the liners should be used, *i.e.* that which allows for the growth of the boundary layer and that for higher Mach number, a liner divergence of $0{\cdot}5°$ was beneficial.

An overall variation of Mach number of $\pm 0{\cdot}005$ was finally obtained in the working section throughout the transonic range.

(*iii*) MEASUREMENT OF PITCH IN THE AIR STREAM

Towards the end of the calibration of the working section, a survey was made of the flow inclination in pitch using a wedge type pitchmeter. This showed that in the area occupied by the wing there were minor variations of pitch of about $\pm 0{\cdot}15°$ over the range of Mach number. There was, however, a slightly larger variation from the wing to the tailplane position which amounted to about $\pm 0{\cdot}2°$, depending on the Mach number. Before applying any correction to tail

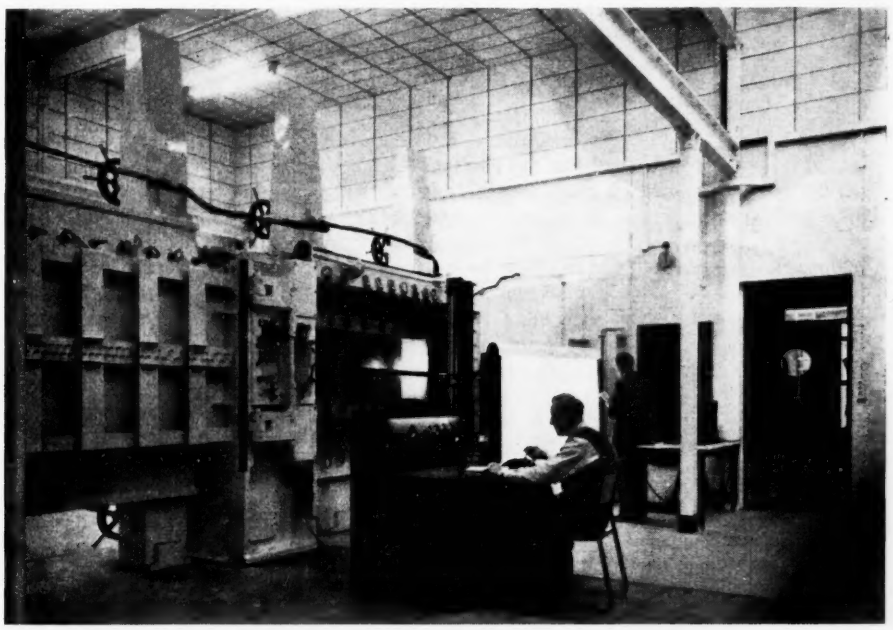

FIGURE 3. General layout of the control room.

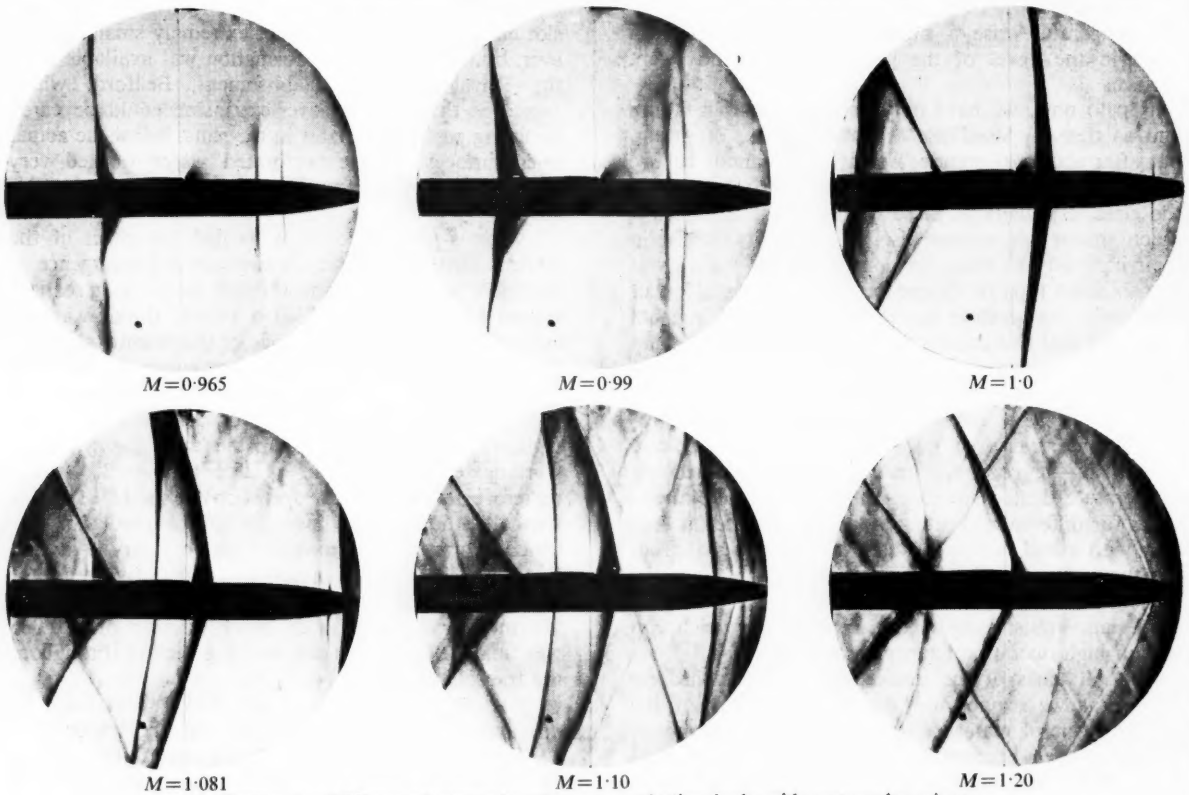

FIGURE 4. Schlieren photographs of pressure plotting body with rectangular wings.

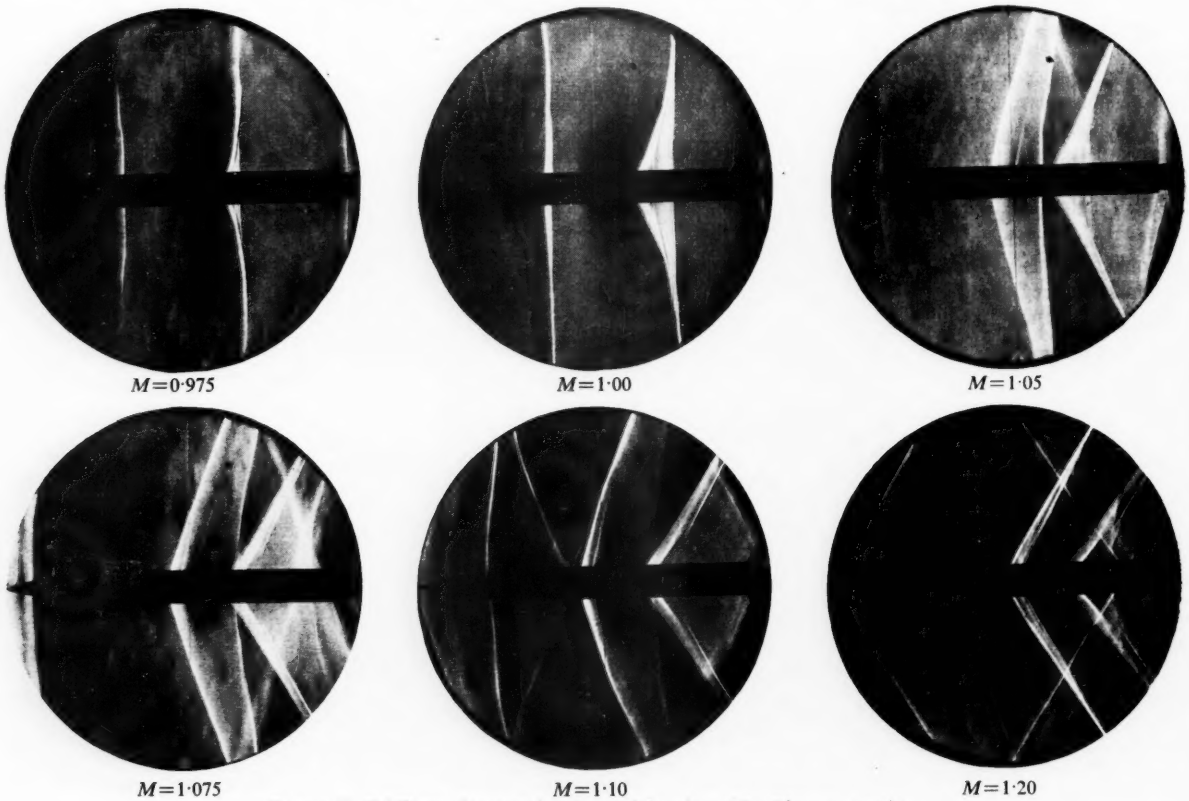

FIGURE 5. Schlieren photographs of a straight wing with a biconvex section.

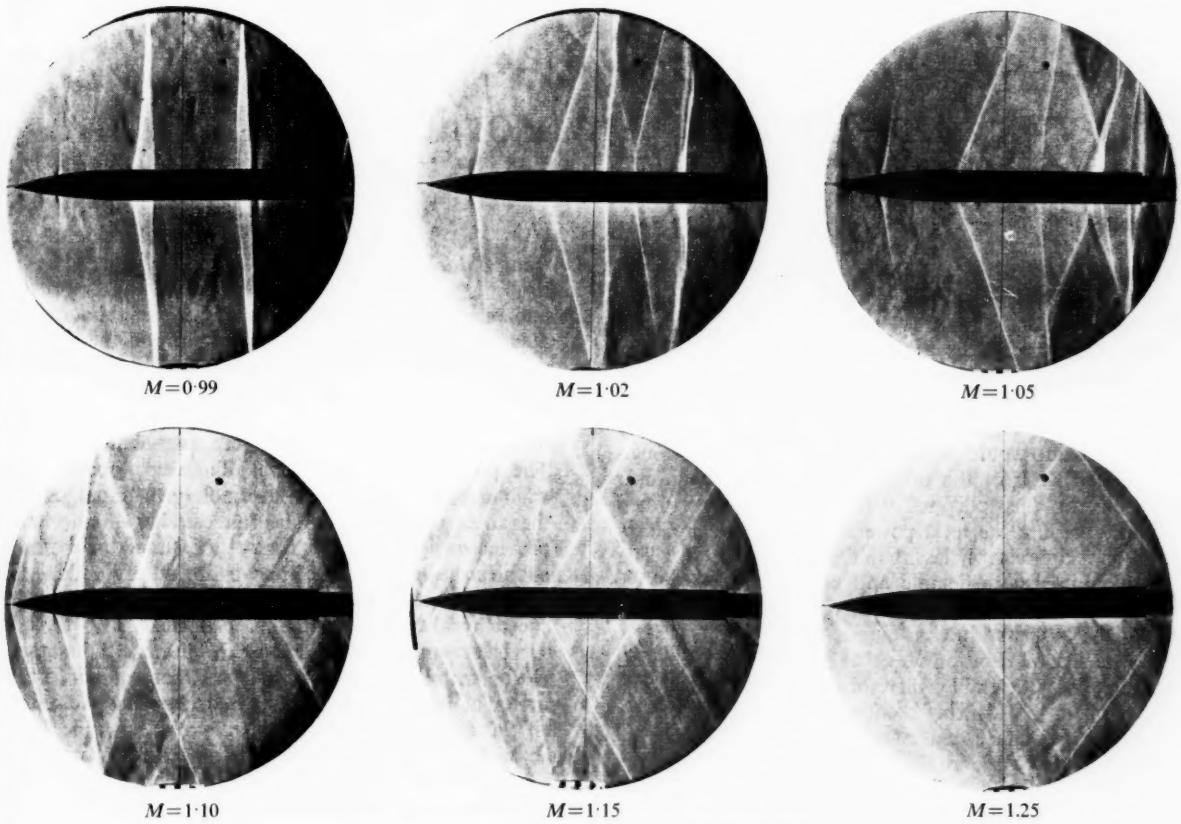

FIGURE 6. Schlieren photographs of a 60° swept wing.

settings to allow for this variation, it is proposed to repeat some of these results with the plenum tanks linked.

This brings me to the last problem we had to solve. During tests made previously on a research model of 0·5 per cent blockage it was found that an appreciable differential pressure existed between the two plenum chambers and that this change of pressure was a function of the wing loading. It was suspected that this differential pressure would induce an upwash near the model and that the lift curve slopes measured would be inaccurate. Several tests were repeated with the two plenum chambers linked by two large ducts which equalised the pressure in both tanks and the results have shown that the previously measured lift curve slopes were only in error by a very small amount.

(iv) SHOCK WAVE REFLECTION

It is now an accepted fact that slotted working sections, especially those incorporating a small number of slots, do not effectively reduce the reflection from either shock or expansion waves. This means that over a certain range of transonic speeds, the model will be hit by reflection waves which, of course, the full scale aircraft will not meet. As these waves can also cause separation if the local boundary layer conditions are favourable, it is important to know the effects of these reflected waves and, in the de Havilland tunnel, the model is constantly watched with the aid of schlieren apparatus. Ample photographs are taken during the tests and these are invaluable when analysing the results of the forces measurements.

Typical sample photographs are shown in Figs. 4, 5 and 6 which were taken with three different fuselage-wing combinations but as the wing span was horizontal when the photographs were taken, the position of the wing can only be seen by the leading and trailing edge shock waves.

In each set of photographs the rearward movement of the body shock can be seen as the speed is increased until it merges with the wing leading edge shock. The intersection of the fuselage bow wave and of the wing shock with the schlieren window can also be seen together with the small shock wave caused by the transition wire fitted to the fuselage; this clearly shows the difficulties experienced when fixing the transition on the fuselage without creating other disturbances which would not occur on the full scale aircraft.

The reflection of the various shock waves on the horizontal liners can be seen very clearly if one remembers that the level of the liners is slightly above and below the vertical range of the photographs.

Design and Operational Problems of the Electrically-Driven Transonic Wind Tunnel

by

R. HILLS, B.A., A.F.R.Ae.S.

(*Aircraft Research Association Limited*)

Introduction

The Aircraft Research Association's tunnel was designed from the start as a transonic tunnel. Two important decisions were taken at the beginning of the design of this tunnel. First, that a separate auxiliary suction plant would be used to suck air out through ventilated walls of the tunnel instead of using diffuser suction for this purpose. This decision was taken mainly because the total power required with auxiliary suction is appreciably less than for diffuser suction. The plant is also more flexible and provides, for example, the possibility of running a smaller supersonic tunnel from the auxiliary suction plant.

The second decision was that a Mach number of at least 1·3 was desirable, therefore a flexible nozzle was provided in the working section so that it could be set to give supersonic speeds and so avoid the necessity for having a very large auxiliary suction plant.

Main Tunnel Circuit and Drive

The main tunnel circuit is shown in Fig. 1. It is of conventional design with the fan downstream of the second corner and a large rectangular cooler immediately upstream of the third corner. The tunnel can only be operated at pressures near to atmospheric, the structure being designed to allow the pressure to be varied by $\pm 1/5$ atmosphere from this figure. Immediately downstream of the contraction is a single smoothing screen and the contraction ratio is 11 : 1. With this arrangement the flow seems to be very steady in the working section, although no direct measurements of steadiness or turbulence have yet been made.

The fan shown in Fig. 2 consists of two stages mounted on a common shaft, each stage with 20 blades. The pre-rotation vanes have adjustable flaps at their trailing edges which can be remotely operated so that the angle of swirl into the fan can be varied. This provides a very useful control of tunnel speed and enables the tunnel to be run at top fan revolutions for Mach numbers of 0·85 and above with control of tunnel speed by altering the stator flap angle.

The main motor drive is a 25,000 h.p. a.c. induction motor with liquid resistance speed control and a d.c. motor of 1,500 h.p. on the same shaft. This d.c. motor is Ward Leonard controlled and provides a fine speed control on the system. This arrangement works very well and there is no difficulty in holding the fan speed to within the specified accuracy of 0·1 per cent.

Working Section

The working section itself is contained in a large pressure shell and air is sucked out of this shell by an 11-stage axial compressor with a maximum power of 13,700 h.p. After it has gone through the auxiliary suction plant the air is returned back into the main tunnel stream through a slot round all four walls of the

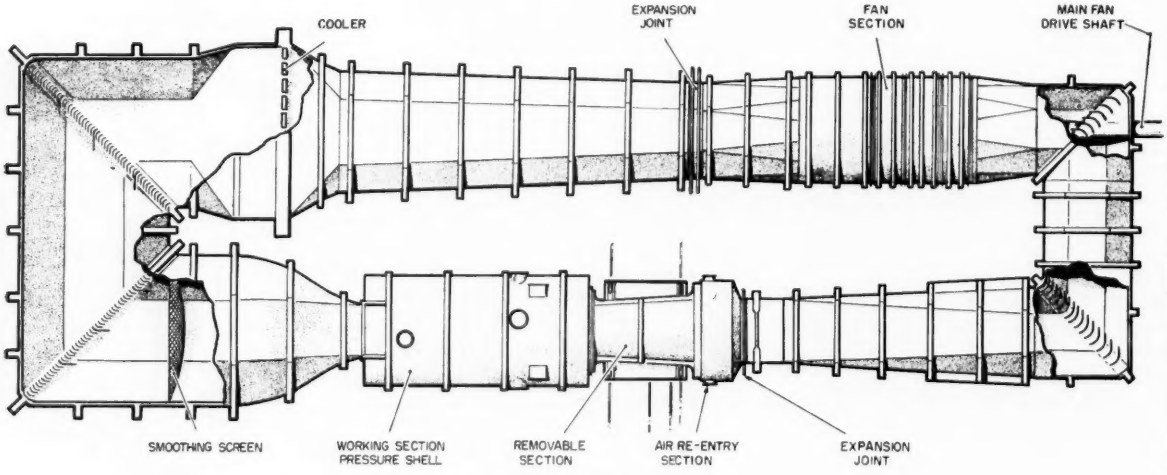

FIGURE 1. Tunnel circuit.

tunnel just downstream of the working section. This serves to improve the efficiency of the first diffuser of the tunnel and reduces the main drive fan power.

The tunnel has been arranged so that models can be taken in or out rapidly, by mounting the models on carts which can be run in or out of the tunnel on rails. Access into the tunnel for the carts is provided by moving the first part of the diffuser immediately downstream of the working section out to one side.

The details of the working section are shown in Fig. 3. The upstream section consists of fixed parallel side walls with the floor and roof adjustable in shape to form the supersonic nozzle shapes. Immediately downstream of this flexible nozzle there is a 12 ft. long section in which all four walls consist of perforated plates. To reduce the effects of a shock wave due to the step at the junction between the perforated plates and the flexible nozzle and walls, the joint takes the form of a serration with teeth at 60°.

In this section the top and bottom walls are fixed while the side walls can be flexed about their upstream ends by approximately $\pm 1°$. Since at any given Mach number more air will be sucked through the walls when they are converged, this gives a useful control of the boundary layer thickness on the walls of the working section. By using this effect the best conditions for the elimination of shock wave reflections from the walls at low supersonic speeds can be obtained.

The main reason for using perforated walls on this tunnel, in contrast to the vast majority of other transonic tunnels, is that it was thought that there was a much greater chance of avoiding shock wave reflections than with tunnels with longitudinal slots. This is a question which has not been fully resolved and it has proved difficult to do any reliable test work in the small perforated tunnels which have been available in the past. We think that with the A.R.A. tunnel we can get some really reliable results to investigate this problem.

Downstream of the perforated section is the model cart and the side walls are again hinged, this time about their rear ends so that they can be adjusted to give either a zero gap or a small gap at the end of the perforated wall. By using the gap at this point it has, in fact, proved possible to run the tunnel with diffuser suction and, with the walls diverged, a Mach number of 1·1 can be achieved without any auxiliary suction.

Flexible Nozzle

The detailed mechanism of the flexible nozzle is of some interest as it is a novel arrangement which we do not think has been used before. All four walls in this area are built up from strips of cold drawn steel 4 in. wide by $\frac{3}{8}$ in. thick on the top and bottom, 5 in. wide by $\frac{1}{2}$ in. thick on the sides. These are fixed by countersunk bolts at approximately $13\frac{1}{2}$ in. intervals to cross members; the countersunk holes are filled with Araldite to present a smooth surface to the wind stream. The longitudinal joints between the individual planks are sealed with rubber strips to prevent any leakage of air between the planks. Since the surface of the planks is equivalent in flatness to a good machined surface, this method of constructing the walls avoids the cost of

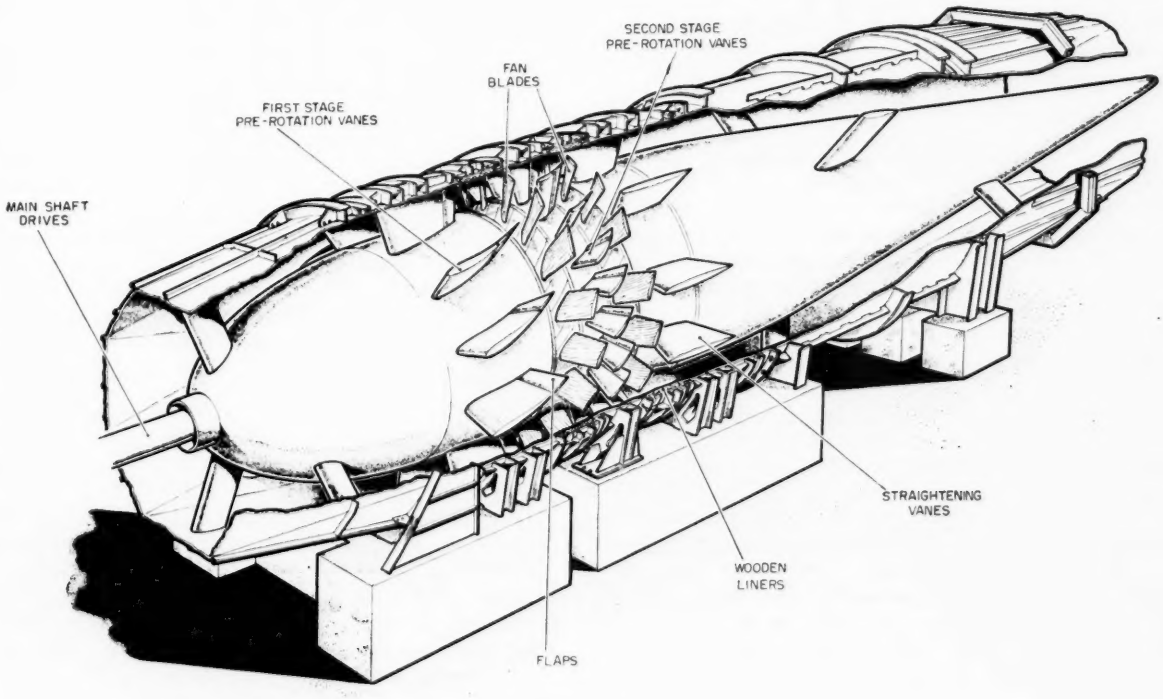

FIGURE 2. Fan.

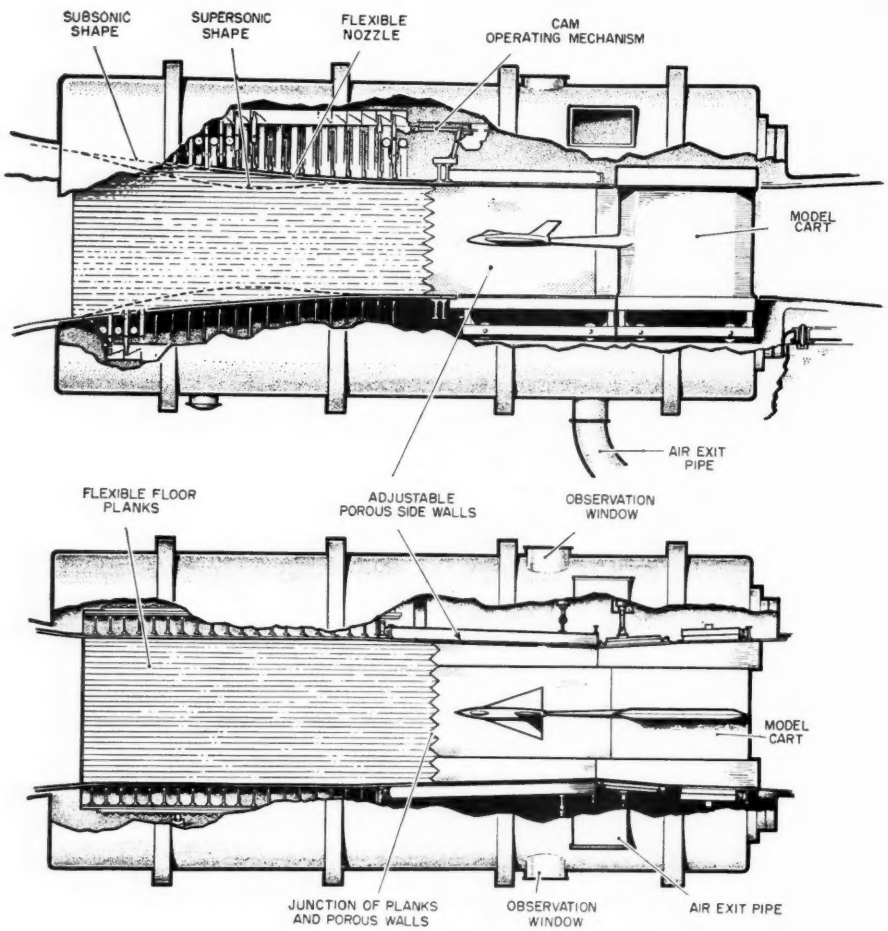

FIGURE 3. Working section.

expensive machining of a single plate which would be 25 ft. long by 8 ft. or 9 ft. wide.

The adjustment in shape of top and bottom plates is obtained by fixing them at every 13½ in. along the plates to cross members by a hinge near the plate, these cross members are then fixed by pivots to vertical slides on each side, as shown in Fig. 4. The vertical position of each of the slides is determined by a cam which is cut in a series of steps. Under working conditions the vertical slide is pulled on to the cam by a rope round a pulley which is fixed to the slide. To alter the vertical position of the slide, and hence the shape of the plate, the slide can be pulled off the cam by another rope and pulley. The cam can then be moved along one step and the slide pulled back on to the cam. All the cams on one side are joined together so that they are all moved at the same time and similarly, a single rope operates all the pulleys moving in one direction on one side. Tension in the rope is maintained at about 4,000 lb. by a slipring motor specially designed to produce a high torque at zero speed.

Nine different steps are provided on each cam, giving nine different shapes of nozzle, $M = 1{\cdot}0$ or less, 1·15, 1·2, 1·25, 1·3, and so on to 1·6. The higher speeds are provided to allow for future development of the tunnel. The whole arrangement is controlled from push buttons on the control desk in the Observation Room. Intermediate Mach numbers can be obtained by varying the amount of suction through the perforated walls.

Results

Some preliminary results from the tunnel are shown in the next figures. Fig. 5 shows the auxiliary suction mass flow required for various Mach numbers. The figure of one per cent mass flow at $M = 1$ is appreciably lower than obtained on other tunnels, partly due to the fairly short length of perforated section and partly to a judicious choice of the hole size and plate thickness.

The great advantage of using a flexible nozzle to reduce the amount of suction required is evident. The line marked as M nozzle $= M$ plenum is the minimum Mach number which can be obtained for a given nozzle shape. If the suction is reduced below this value the flow collapses, just as it does in a normal supersonic

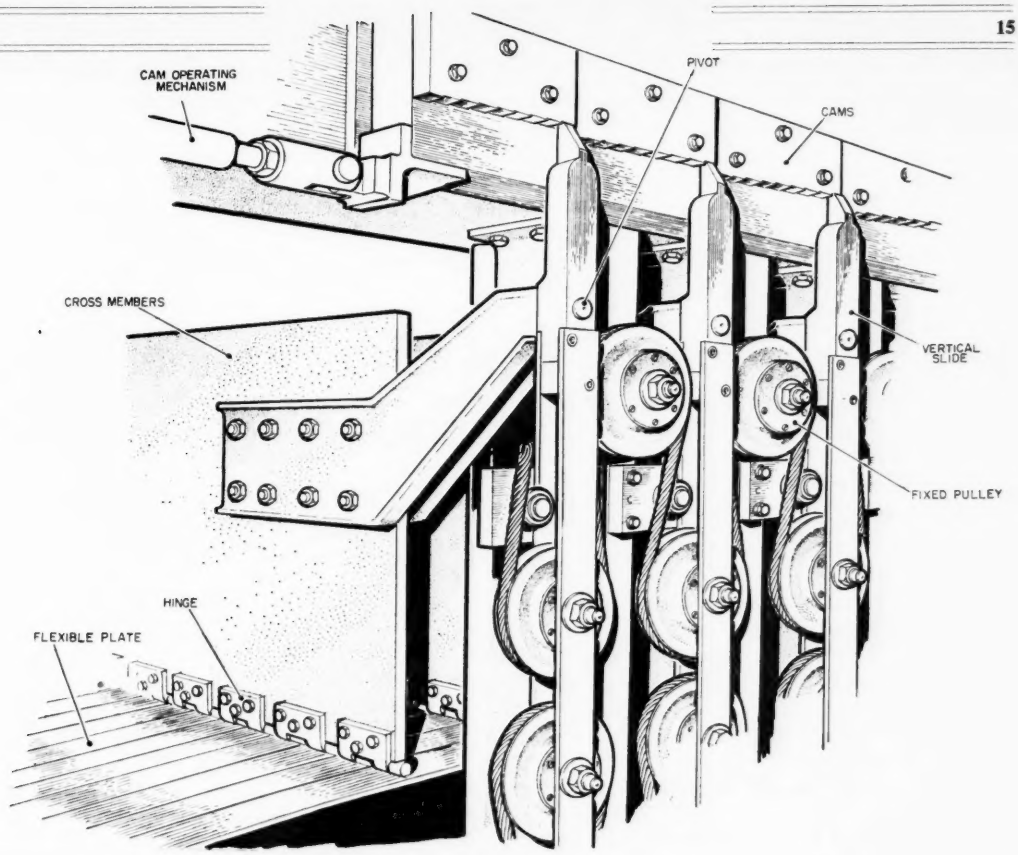

FIGURE 4. Flexible nozzle.

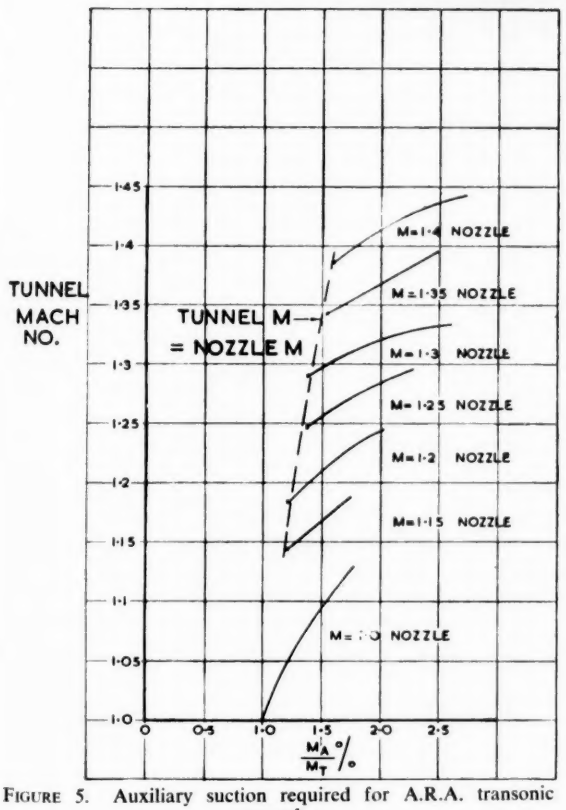

FIGURE 5. Auxiliary suction required for A.R.A. transonic tunnel.

tunnel with the shock wave moving up to the beginning of the perforated section.

It can also be seen that a Mach number over 1·4 has been achieved. There is still some main fan power and suction power in hand and it seems probable that in the empty tunnel a Mach number above this could be reached. Allowing, however, for the effects of a model, about 1·4 is the most useful working limit.

Figure 6 shows a typical set of centreline distributions in the working sections. At $M=1$ or below, the Mach number distribution is very even and most of the kinks are probably due to imperfections in the measuring tube, rather than flow imperfections. Above $M=1$, however, it can be seen that there are a number of

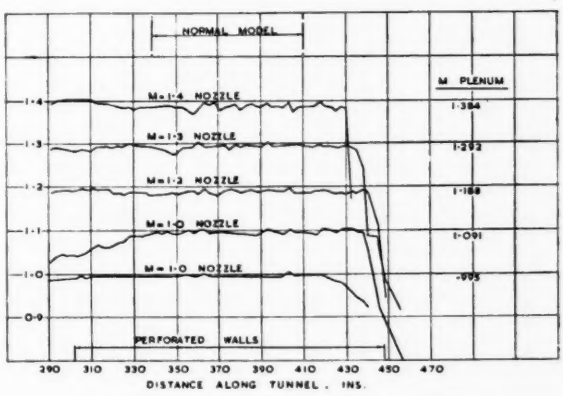

FIGURE 6. Centre line velocity distribution.

disturbances which become rather worse as the Mach number is increased. Some efforts are still being made to eliminate these. At $M = 1{\cdot}4$ in particular there is a fairly large disturbance which appears to be associated with the joint between the flexible nozzle and the perforated wall, mentioned earlier, which consists of teeth inclined at 60°.

So far the tunnel has proved to be very steady in operation and no difficulty has been experienced in doing calibrations or model tests.

Some Aspects of Transonic Tunnel Operation in Industry

by

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(*English Electric Co. Ltd.*)

Introduction

Early in 1950 it was decided to modify experimentally the original jet-driven tunnel of the English Electric Company—Aircraft Division, from a subsonic to a transonic tunnel. This was inspired by the work of Ward and Wright[1] at Langley Field. As little was known then of the pressure ratio to operate a transonic working section, the working section area was reduced from a 3·8 square foot subsonic section which could be choked, to one square foot. The working section and diffuser for this modification were quickly manufactured from wood. In addition to the new working section and diffuser, a steel fairing was added to reduce the cross sectional area of the mixing length and so match the smaller induced mass flow. This transonic tunnel first ran in July 1950 and from the outset transonic flow was generated in the working section, although initially the distribution was poor. Various slotted and perforated walls were tried, and all were completely surrounded by a plenum chamber of two-foot square section and three feet long.

It was found that the pressure ratio required for the slotted walls was considerably lower than for the perforated walls and all subsequent development work has been done on the former.

Each wall consisted of ten T-shaped light alloy extrusions, the slot width tapered from zero at the entrance to an open to total area ratio of 1/6 in

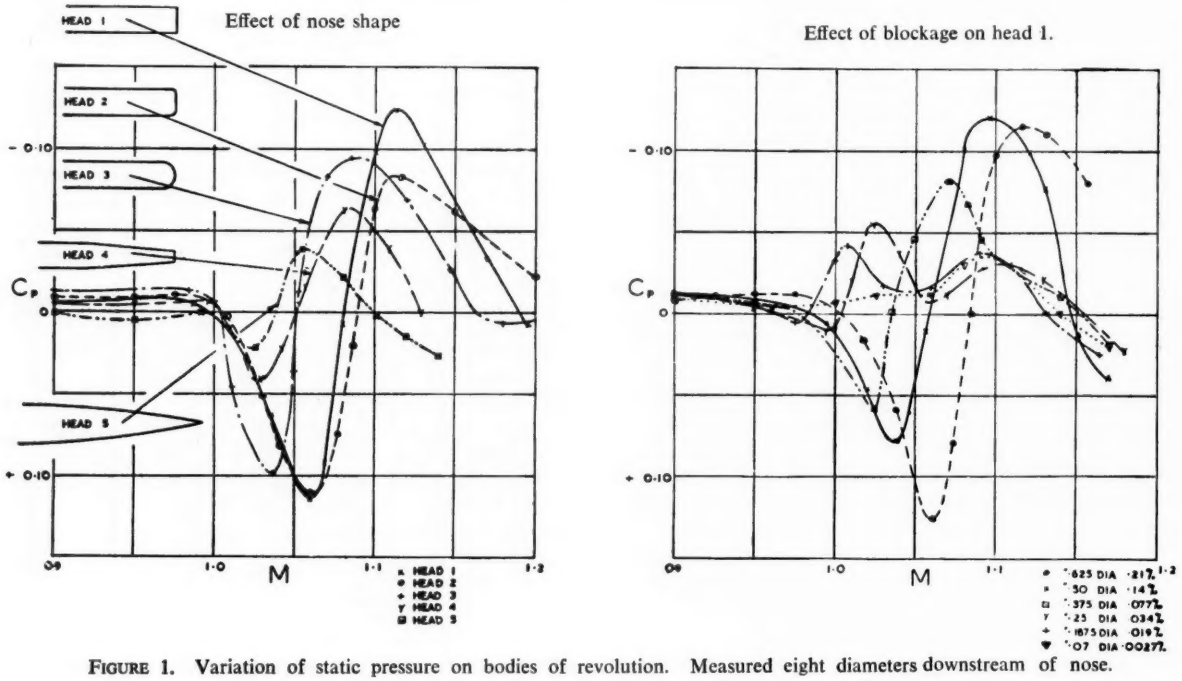

FIGURE 1. Variation of static pressure on bodies of revolution. Measured eight diameters downstream of nose.

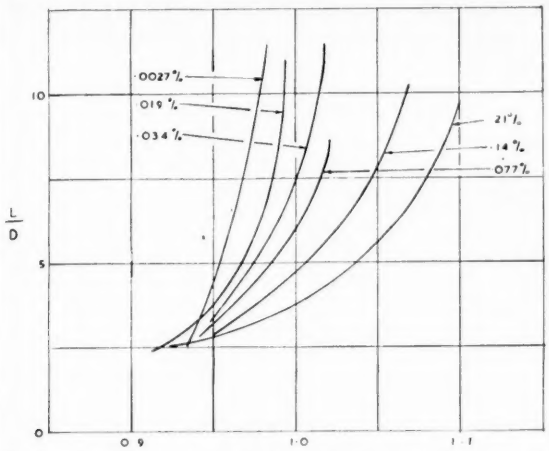

FIGURE 2(*a*). Effect of model blockage on position of normal shock.

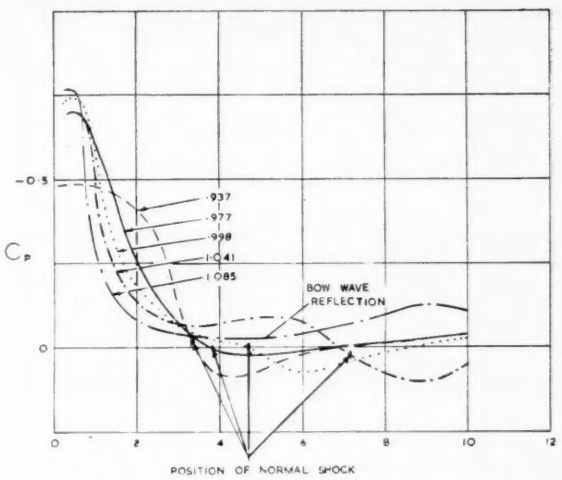

FIGURE 2(*b*). Effect of Mach number on pressure distribution (bluff nosed body of revolution).

length equal to the tunnel length. Numerous modifications were tried; convergence and divergence of the walls, various shapes of effuser fairing, differing amounts of exit area from the slots, and boundary layer control in the diffuser. The maximum Mach number was increased to 1·22, and although there was still an over-expansion through the entry to the slots at the top Mach number the variation in M was still $\pm 0{\cdot}01$. The distribution of Mach number was considerably better at lower Mach numbers. This performance was considered adequate for test work.

Body of Revolution, Investigation and Interference Effects

Initially, little concern was given to possible interference effects, the limited information available suggested that these would be small. However, late in 1951 a five-sixths scale model of a Mark IX Pitot Static head was tested in the tunnel. This test showed a considerable variation in the static pressure through the transonic speed range (*see* Fig. 1, head No. 3). Schlieren windows were added to the plenum chamber and it was immediately apparent that the static pressure behaviour could be attributed to a shock wave phenomena. At a Mach number of about 0·9 a normal shock occurred at the rear of the supersonic region associated with the expansion round the nose of the head. As the Mach number was increased, this shock increased in intensity and moved slowly downstream. It crossed the static vents at $M = 1{\cdot}05$ and caused large variations in the measured pressure coefficient. A short distance downstream of the shock, the Mach number returned almost to free stream value, so that the shock could not be regarded as the final tunnel shock. No similar pressure variations were revealed in the available flight test data. Thus the presence of the shock at $M > 1$ could not be explained as other than due to tunnel interference.

A very limited programme was undertaken to investigate the shock wave phenomena. Several simple models were made to find the effect of nose shape and of blockage area. The majority of these models were based on the Pitot Static, that is, they had one static measuring station situated eight body diameters back from the nose, although a few were pressure plotted over their length. Some of the results are shown in Fig. 1, where the variation of static pressure is shown plotted against Mach number. It will be seen that each model exhibited a similar behaviour. In the region of $M = 1{\cdot}0$ the pressure coefficient became positive, then on increasing M there was a sudden "kick" and the pressure became negative before returning to a value nearer free stream. These "kicks" were caused by the passage of a near normal shock. There was a considerable increase in the strength of a shock due to it occurring at a higher Mach number with increase in model bluffness and also, model size. The effect of model size, Fig. 1(*b*), was most marked, this being further illustrated by Fig. 2(*a*) where the distance of the normal shock, measured downstream from the nose, divided by the body diameter, is shown plotted against free stream Mach number. The speed at which the normal shock moved rearwards with increase in Mach number was so reduced by increase in model size that for the largest model (0·21 per cent blockage), the normal shock was still present on the body at $M = 1{\cdot}1$. In contrast, for the smallest model (0·002 per cent) the shock had reached the rear of the model before $M = 1{\cdot}0$.

A more detailed picture of the behaviour of the shock system is given by Fig. 2(*b*) which shows the variation of the static pressure distribution with Mach number.

Below a test Mach number of 0·94 there was a region of low pressure which extended from the nose to approximately two body diameters downstream; this region was shown by the schlieren to be terminated by a normal shock. The increase in pressure due to the shock was softened, presumably due to the separated boundary layer, so that it was spread over a length equivalent to almost two body diameters. The pressure

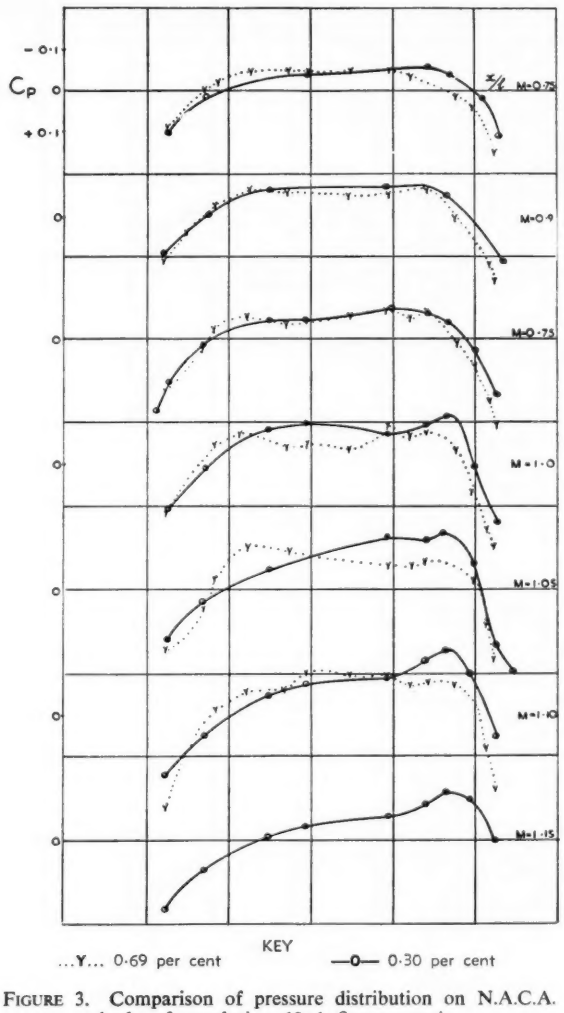

FIGURE 3. Comparison of pressure distribution on N.A.C.A. body of revolution 12:1 fineness ratio.

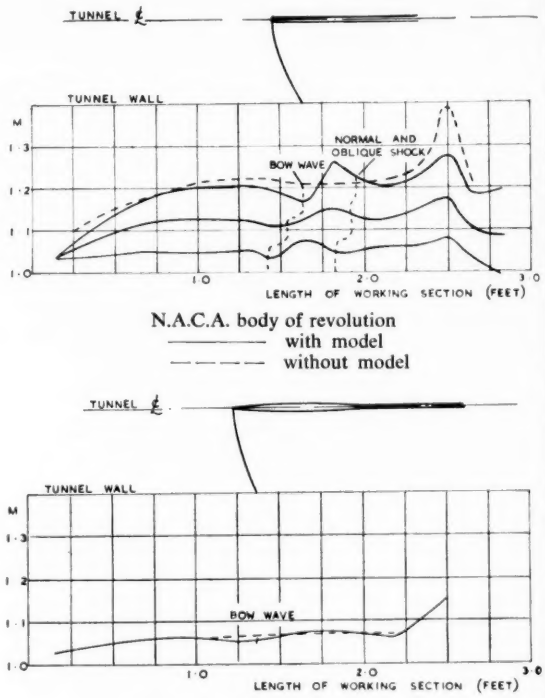

FIGURE 5. Pitot head type 1, $\alpha=0°$.
Effect of flow disturbances on Mach number measured at the the wall.

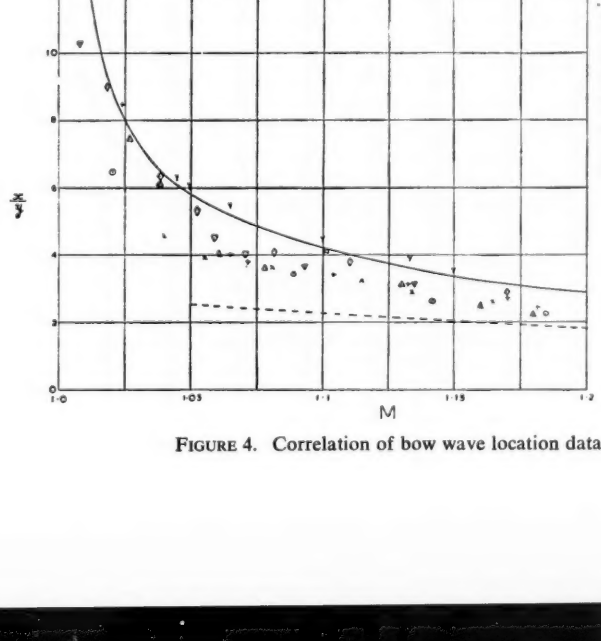

FIGURE 4. Correlation of bow wave location data.

had returned to the free stream value after a total of six diameters ($X/D=6·0$). Between $M=0·94$ and $M=0·98$ there was an abrupt change in the pressure distribution near the nose which coincided with the formation of the separation bubble and an oblique shock, the value of the peak negative Cp increased from $-0·5$ to $-0·75$ and the extent of the peak was confined to a length of aproximately one diameter. The pressure distribution in the region of the separation bubble remained practically independent of Mach number throughout the supersonic speed range covered in the tests. After the oblique shock the static pressure remained slightly lower than free stream up to the normal shock. As the free stream Mach number was increased, this region of negative Cp and the normal shock moved downstream and at $M=1·085$ and $M=1·02$ was downstream of the observed reflection of the bow-wave. It would therefore seem possible that the bow-wave was reflected as an expansion wave,

in addition to the weak shock observed on the schlieren. This might be expected for the working section has a rather large open area, from which the incident shock would be reflected partly as an expansion.

All the previous results were obtained on bodies which were characterised by having shocks forming at subsonic speeds near the nose, and in front of a long length of body with very little pressure gradient. To investigate the flow behaviour on bodies with a different type of pressure distribution, two 12 : 1 fineness ratio bodies of revolution were tested, Fig. 3.

Schlieren observations showed that a strong normal shock first appeared at $x/l=0{\cdot}7$ and it moved rapidly downstream to the body-sting junction with increase in Mach number. Consequently, the tunnel wall interference on the behaviour of this normal shock was minimised. The larger model results agreed with the smaller results up to $M=0{\cdot}9$, suggesting that subsonic blockage effects were quite small.

At higher Mach numbers, however, with the larger model, the tendency was towards increased suctions near the fore part of the body and higher pressures near the tail. This would suggest that there was an appreciable static pressure gradient in the tunnel due to the presence of the larger model, which would accordingly experience a thrust. Apart from this "buoyancy" effect it would appear that there was no strong interference effect of the walls on the flow pattern around these aerodynamically clean models.

Bow Wave Position

The bow wave positions for the various models are shown in Fig. 4, plotted as distance of the bow-wave from the sonic point divided by the semi-ordinate of the model at the sonic point against Mach number. For all bodies except the N.A.C.A. bodies of revolution, the sonic point has been assumed to occur at the shoulder of the model. Two theoretical curves obtained by the geometric method and the continuity method[2] are shown for comparison. The experimental points show better agreement with the geometric method, especially at the lowest Mach numbers. It is also at the lowest supersonic Mach numbers that the results on the $\frac{5}{8}$ in. diameter and $\frac{1}{2}$ in. diameter bodies diverge from the general trend, the bow waves appearing closer to the model. As these were the strongest bow waves it is probable that the slotted walls were having a slight effect on bow wave position.

Variation of Pressure at the Tunnel Walls

The results just discussed have all been based on measurements at the model. The effects of a model on the pressure measured along a side wall are shown in Fig. 5. Only a limited number of wall pressure points were available situated along the centre line of one of the slots, but they show the general trends. It will be observed in Fig. 5(*a*) that, although there are two strong incident shocks reaching the wall, *i.e.* the bow wave and the normal shock, the mean pressure at the wall remains constant and approximately equal to the free stream static pressure, which approximates to the plenum chamber pressure.

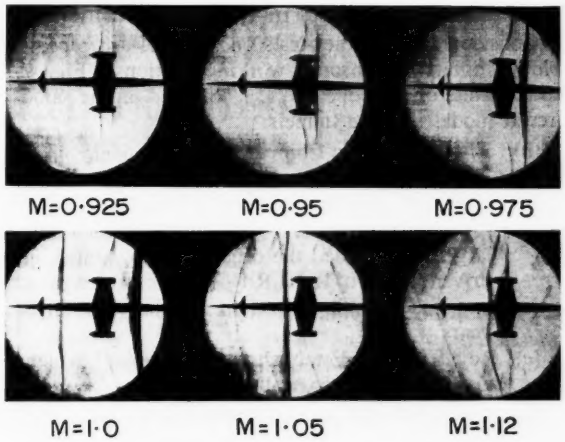

FIGURE 6. Shock wave development in a slotted wall working section. Blockage = 0·5 per cent.

Slotted Wall Interference

Studying these tests it is apparent that a slotted wall tunnel can considerably modify the flow pattern around even a very small body. The interference effects increase with both model bluffness and blockage, and are worst with a model which develops a strong shock system near the nose at subsonic speed.

Adopting a simple minded approach to these phenomena, we believe that they are due to the fact that the tunnel boundary has a considerable percentage of open area, which cannot support a large pressure differential. We have found that when an expansion region terminated by a shock reaches the tunnel wall, the strength of the shock was such that the mean of the pressures before and after approximates to plenum chamber pressure. So that this balance can be achieved, the flow pattern may be considerably modified, extending from the wall to the body. Consequently, interference effects may be large when strong pressure disturbances from a body reach the tunnel walls and care must be exercised in interpreting the results obtained. I have said nothing about shock reflections as this subject has been covered by previous speakers.

A typical set of schlieren photographs on an aircraft model is shown in Fig. 6. The model does not represent any particular design, but is similar to the futuristic designs which can be seen in the popular press. Strong shocks from the nacelles reach the tunnel walls at $M=0{\cdot}975$ and the large interference effects must begin then. It will be noticed that there is a strong normal shock on the model at $M=1{\cdot}0$ which could have an adverse effect on the fin.

It must be stressed that all this data was obtained in a slotted wall tunnel with a rather large open to total area ratio, and the results might have been different for a perforated wall tunnel, or one with lower porosity.

Comparison of Results in Different Tunnels

Consider now whether tests on a model are repeatable in different transonic working sections. Fig. 7 shows results of lift slope and C_{D_0} obtained on the same aircraft model in three tunnels:

(*a*) English Electric Company 12 in. tunnel.

(*b*) English Electric Company 18 in. tunnel, which is slotted on only two sides and has an open to total area of 1/12. This tunnel, which has largely superseded the original 12 in. tunnel, is driven by two Rolls-Royce Nene jet engines and has been in operation for the past three years.

(*c*) Royal Aircraft Establishment 3 ft. tunnel, which also had only two walls slotted.

The blockages of the model in these tunnels was respectively 2·6 per cent, 0·75 per cent and 0·2 per cent.

It will be seen that the lift slope in tunnel (*a*) agrees very well with that obtained from tunnel (*c*) in which it can be assumed that the model has negligible blockage.

The lift slope from tunnel (*b*) however, is considerably lower. This is almost certainly due to the fact that in this tunnel an appreciable flow angularity was permitted, as the plenum chambers were unconnected and were at slightly different pressures.

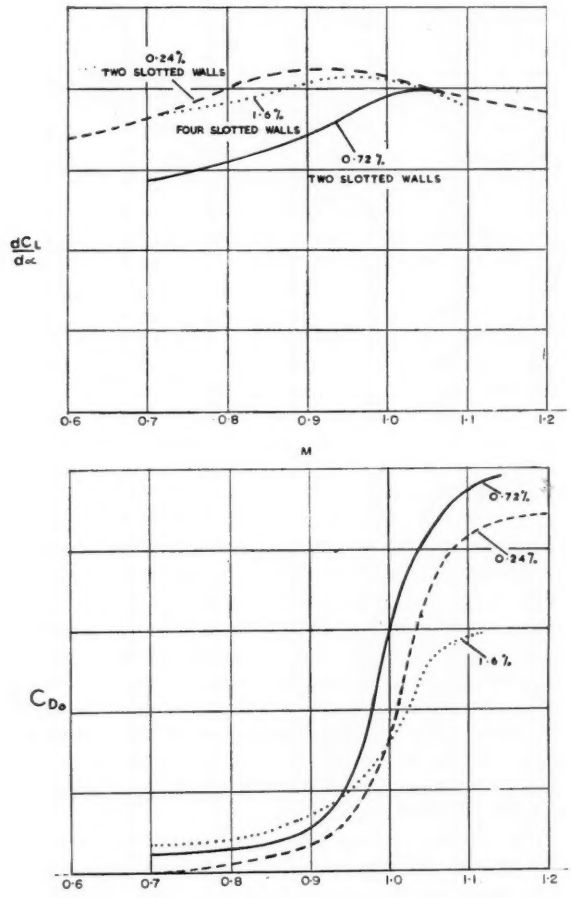

FIGURE 7. Effect of blockage on model results.

On the other hand, accurate lift slopes were obtained with a very large model (2·6 per cent blockage) when the plenum chamber extended completely round the working section.

The effect of blockage on drag is different. Here the results in the two larger tunnels are similar but the tests at 2·6 per cent blockage show a smaller transonic drag rise. It is thought that this is due to the static pressure gradient noticed on the earlier body of revolution tests, *i.e.* reduced pressures at the nose and higher pressures near the rear of the model giving an effective thrust.

Comparison With Flight

The ultimate test of the validity of transonic tunnel results is to compare them with flight test results. Security reasons prevent me making any actual comparisons in this paper, but we have been able to compare some tunnel tests with data obtained in flight on the P.1 aircraft. When the tunnel results have been corrected for mass flow and aeroelastic effects, quite good comparisons have been obtained. The general trends and the characteristics of the aircraft in the transonic range have been faithfully predicted by the tunnel, and sufficient correlation has been obtained to encourage the use of slotted wall tunnels in this critical range.

Conclusions

Summarising our experience we can say that it is comparatively easy to develop a tunnel to operate satisfactorily throughout the transonic ranges. It would appear, however, that appreciable interference effects are possible both below and above $M = 1{\cdot}0$. These interference effects are not only a function of model size, but also of model "dirtiness."

There is an obvious need for much more fundamental research work on these interference effects.

Wherever possible we try to observe the following principles:

(*a*) The model should be as small as possible.

(*b*) When strong pressure fields originating at the model reach the walls due to either model "dirtiness," high lift, and so on, care is required in interpreting the results. In this connection, we, like de Havilland, have found schlieren invaluable.

Nevertheless we have found that transonic tunnel results can give a valuable guide to full scale performance.

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2. MOECKEL, W. E. Approximate method for predicting the form and location of detached shock waves ahead of plane or axially symmetric bodies. N.A.C.A. T.N. 1921.

The Assessment of Results obtained in Transonic Wind Tunnels

by

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1. Introduction

The development of wind tunnel sections with partly open walls has made it possible to operate at continuously variable speeds through the transonic range up to low supersonic speeds. Some consideration must be given, however, to the general reliability of the measurements in these tunnels of the aerodynamic characteristics of air vehicles or missiles. A comparative assessment of the possible overall errors in force and pressure measurements in relation to desirable standards of accuracy is, in fact, required. A number of points connected with such an assessment are briefly discussed (mainly on the basis of experience in the 3 ft. tunnel at R.A.E., Bedford) in this paper, which touches on flow uniformity, accuracy of model manufacture, and interference effects, including a comparison between tunnel and free flight model results; illustrative examples of results for an aircraft model, with some full scale flight comparisons, are discussed briefly and a few comments are made on the problems associated with the achievement of representative flow conditions on sting-mounted models at transonic speeds.

2. Uniformity of Flow

The flow distribution in the transonic section of the 3 ft. tunnel at the Royal Aircraft Establishment, Bedford, is taken as typical of the distribution in a slotted wall section in connection with the discussion of uniformity of flow. A brief description of the transonic working section is given first, and this is followed by an outline of the method of flow survey used and of the results obtained.

2.1. TRANSONIC WORKING SECTIONS OF 3 FT. TUNNEL, R.A.E., BEDFORD

The tunnel is of a closed circuit pressurised form driven by two centrifugal compressors in series. It was originally constructed as a supersonic tunnel with a 3 ft. × 3 ft. working section; supersonic speeds are developed by a one-sided nozzle forming the top wall of the working section and a range of top walls with different liners is available to give Mach numbers at 0·1 intervals between $M = 1{\cdot}3$ and 2·0. A fuller description of the tunnel installation is given in Reference 1.

A temporary conversion for transonic operation[2] was made by fitting slotted walls on the vertical sides of the working section (Fig. 1); each side wall has six slots, giving an open area ratio for the whole perimeter of 10·1 per cent. The minimum reduction of the working section was aimed at, and making the plenum chambers behind the slats only 4·6 in. deep allowed a working section of 35 in. × 27 in.; in an unmodified form these proved too shallow as there was a pressure gradient along part of the slotted section associated with secondary flows set up in the plenum chambers by turbulent mixing in the slots; this trouble was almost completely eliminated by the addition of perforated screens behind the slots. The plenum chambers in this arrangement are not connected and, as pointed out in the next section, errors in flow direction may occur because of this.

Recently a new transonic working section has become available with a 3 ft. square working section with slots on all four walls and a surrounding plenum chamber of about one foot depth. There is only one main slot on each wall and half slots at the corners; this arrangement has the advantage that viewing of the model is simplified without reducing the effectiveness of the slotted walls, which in practice is not very sensitive to the number of slots.

2.2. FLOW SURVEY AND CALIBRATION OF 3 FT. TUNNEL TRANSONIC SECTIONS

To determine the longitudinal Mach number distribution in the transonic working sections, flow surveys were made taking measurements of static pressures in a row of static pressure holes along one wall, together with the settling chamber pressure. For more detailed streamwise traverses in the test section a long and very slender tapered probe with a Pitot or static head was used[2]. The Pitot head was of 3 mm. diameter hypodermic tubing cut off square; two static heads were used, and one with pressure orifices on the surface and 1·5 in. from the tip of a 3° cone was found more suitable near sonic speeds than the more conventional cone-cylinder type.

Pitot traverses along the temporary transonic section showed the total head to be constant and equal to the settling chamber pressure and static traverses were made for a range of Mach numbers with the results shown in Fig. 2. These show a small streamwise Mach number gradient at subsonic speeds, fluctuations of $\pm 0{\cdot}004$ at Mach numbers up to 1·05, and fluctuations of as much as 0·008 between 1·05 and 1·15. The value of the plenum chamber reference Mach number M_R estimated from the static pressure in the plenum chambers, is also indicated in Fig. 2; the difference of M_R from the main stream Mach number was $0{\cdot}001 \pm 0{\cdot}002$. The difference

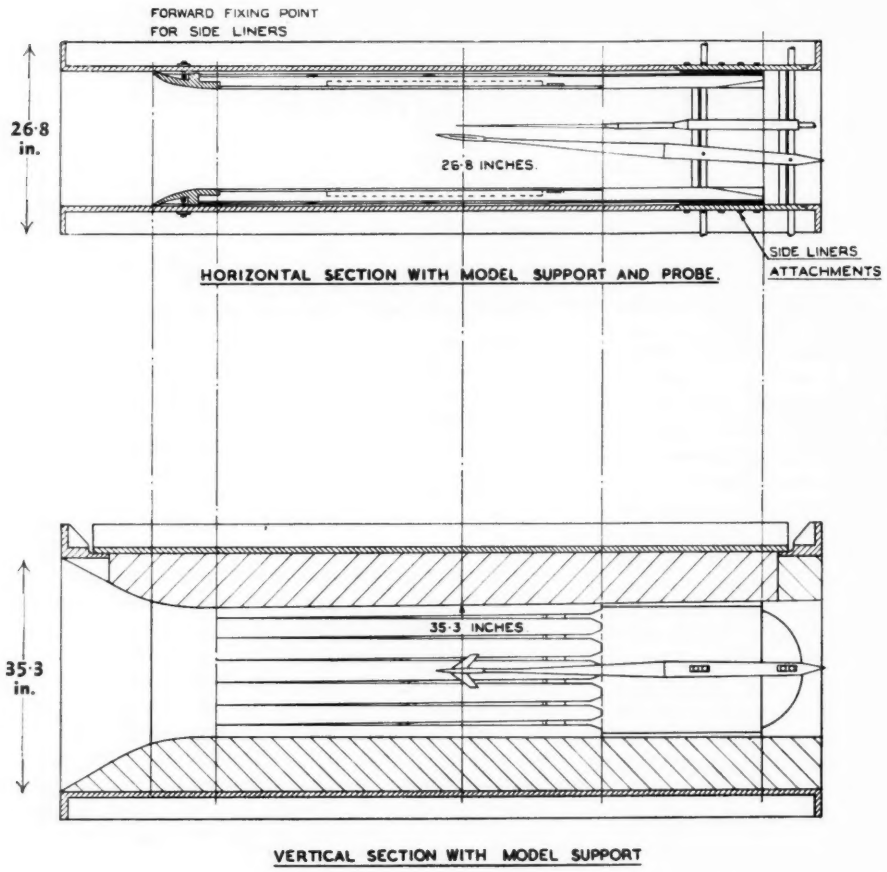

FIGURE 1. Temporary transonic arrangement in 3 ft. tunnel with slotted side liners.

between the reference static pressures in the two plenum chambers was within the limits of experimental error, but they could differ by a significant amount ($\pm 0{\cdot}003$) in Mach number when a large model at incidence was present; this difference is probably accompanied by an induced upwash ahead of the model, even at supersonic main stream speeds.

The flow direction in the empty working section was determined at only one point near the testing position, showing the flow to be inclined $0{\cdot}1°$ approximately to the slotted walls; more extensive calibration would be required to determine whether this is a local flow deviation, or due to a systematic deviation over the testing region for which allowance can be made.

Preliminary calibration has also been made of the new transonic section and this shows the Mach number variation to be less than 0·003 up to $M = 1{\cdot}02$ and not worse than $\pm 0{\cdot}005$ up to $M = 1{\cdot}25$. The angularities of the flow have not yet been measured.

2.3. REQUIRED STANDARD OF FLOW UNIFORMITY

The standard of flow uniformity required depends on the degree of accuracy aimed at in the measurement of aerodynamic quantities, including forces, moments and surface pressures. The overall accuracy achieved depends in addition, of course, on other factors, the main one (apart from instrument sources) probably being model accuracy; it is reasonable to assume that the probable errors arising from these two main sources should be roughly equivalent.

Standards of measurement for transonic conditions can be linked to those defined for supersonic conditions by Morris and Winter[3]. In their method of analysis acceptable limits of random variations in the flow or model shape are assessed by their effects on local surface pressures; systematic flow variations, such as a pressure or velocity gradient, are limited by their effect on overall forces and moments. Thus it has been shown that if the pressure coefficient C_p is to be determined to within $\pm 0{\cdot}005$, the acceptable flow calibration fluctuations at $M = 1{\cdot}4$ are less than $\delta M = \pm 0{\cdot}0025$ and angular deviations less than $\pm 0{\cdot}07°$; a higher standard may be necessary for some research investigations. Recent approximate estimates indicate that the acceptable speed fluctuations are rather smaller at slightly lower supersonic Mach numbers, but for sonic conditions ($M = 1$) are nearly twice those at $M = 1{\cdot}4$.

Errors in force measurements should, in general, be less than one per cent of the maximum in the range considered. At low supersonic speeds, around $M = 1{\cdot}4$, the Mach number gradient per model length for a typical aircraft configuration must be less than 0·4 per cent of nominal Mach number if the buoyancy drag is not to exceed one per cent of the maximum drag; at transonic

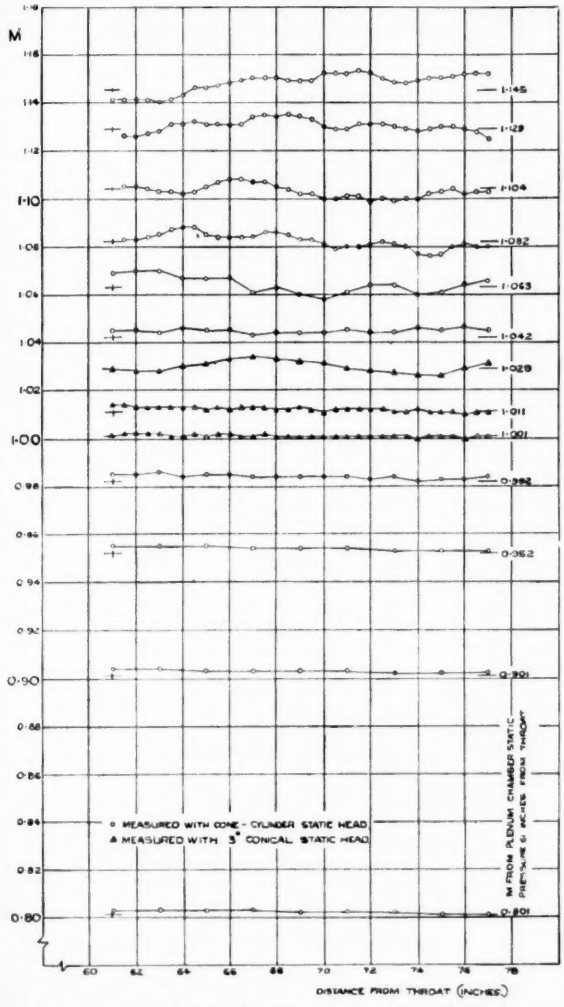

FIGURE 2. Longitudinal Mach number distributions.

speeds near $M = 1{\cdot}0$, the acceptable speed gradient is about twice the above value. Accuracy in speed is particularly important in the region of the drag rise, but drag as an absolute function of speed may be difficult to achieve there, and only the extent of the drag rise may be accurately determined.

On the stability side, the probable requirement is for errors in pitching moment not to exceed a value corresponding to 0·1 degree in tail setting to trim; at $M = 1{\cdot}4$, the acceptable flow deviation is therefore about 0·07 degree per model length, or a gradient in M of 0·0025 per model length. At sonic speed, variations in M of twice this magnitude are possible for the same effect on tail setting to trim. Neutral point errors, not exceeding one per cent chord, are achieved with the flow standards necessary for the trim case.

Corresponding accuracy in the model will require that lifting surfaces should be set to an accuracy comparable to that for flow angularities; for the scale of model used in the 3 ft tunnel this is just possible because wing setting can only be measured to within 0·05 degree and tail setting rather less accurately. Wing surface ordinates for 3 ft. tunnel models are specified to within $\pm 0{\cdot}0015$ in. There is also a limitation on surface waviness of less than 0·05 degree change in slope over 0·1 in., which is important upstream of pressure orifices. Estimates using second order theory indicate that deviations up to the limits of these tolerances can cause differences in aerodynamic quantities of similar magnitude to those due to the flow deviations already specified. Because of the difficulties of model manufacture, it is important that the standard of accuracy called for should not be higher than necessary; experimental evidence on the effect of model errors is needed but cannot be easily obtained and little has been attempted so far.

The effects of distortion of the model, for example to accommodate a sting balance, are touched on later (Section 5).

3. Wind Tunnel Interference Effects

A review of interference effects for tunnels with partly open walls has recently been made by Maeder and Wood[4]. They indicate that subsonic interference effects for both slotted and perforated walls can be eliminated or evaluated; at supersonic speeds, effective wave cancellation is possible with perforations, but with slots only, interaction of waves reflected from different parts of the walls may result in reduction of wall disturbances. In general, however, both at subsonic and supersonic speeds elimination of only one interference effect can be arranged and possibly only for one speed. At high subsonic speeds, for example, the combination of closed and open area ratios producing zero blockage does not, in general, reduce the upwash effect to zero; for both the 3 ft. tunnel transonic sections the blockage correction is reduced to less than 10 per cent of the solid wall value but the upwash effect is of the open type and about 50 per cent and 20 per cent for the temporary and permanent arrangements respectively. These figures correspond to incidence errors of 0·25 degree and 0·10 degree for a typical model at high lift; the blockage corrections are less than 0·005 in. Mach number for speeds up to about $M = 0{\cdot}95$.

It is the practice in the 3 ft. tunnel transonic section to limit the model length to about 0·4 of the test section width, so that the reflection of the bow shock wave from the walls will be clear of the tail at a Mach number below 1·15. Below 1·05 the effects of reflected waves are generally small, but between 1·05 and 1·15 the effects can be large, particularly with models with tailplanes. In special cases marked interference effects are possible below $M = 1{\cdot}05$. This was shown in Roe's tests on a cylindrical Pitot head, discussed in his paper, and in a parallel series of 3 ft. tunnel tests of rather wider scope, of which the principal results are as follows[5].

Pressure measurements made on the cylindrical part of ogive cylinder bodies of various diameters are summarised in Fig. 3. At the point just past the shoulder ($0{\cdot}4D$) there is reasonable agreement between $\Delta p/H$ for the various models up to $M = 0{\cdot}98$ and also

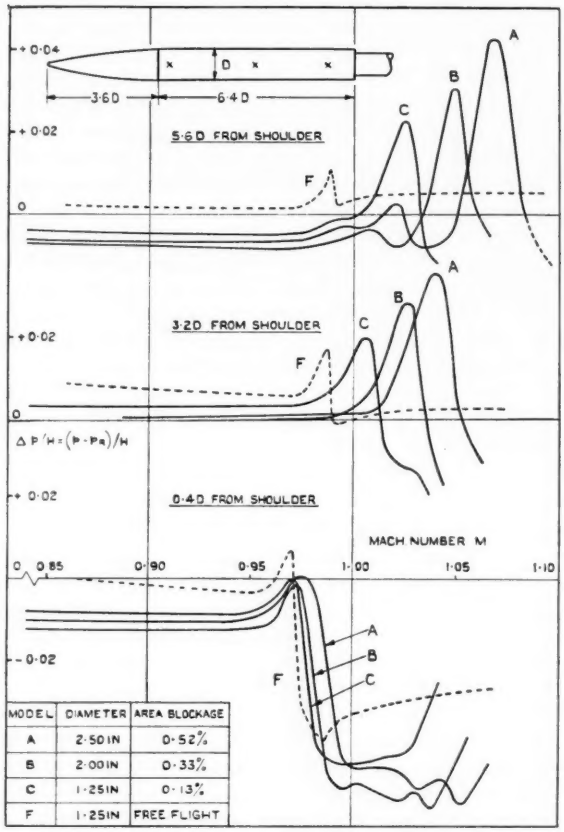

FIGURE 3. Variation of local pressures with Mach number on similar ogive-cylinder bodies in a 35 in. × 27 in. slotted working section and in free flight.

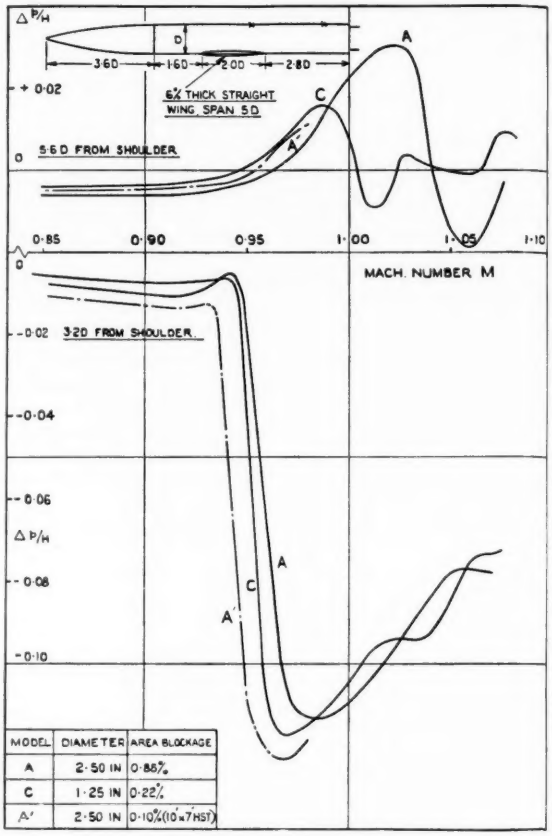

FIGURE 4. Variation of local pressures with Mach number on similar wing and body combinations in a 35 in. × 27 in. slotted working section and a closed 10 ft. × 7 ft. working section.

with results from a free flight test (made at about the same Reynolds number as the tunnel tests); the records all show a steep fall in pressure at the speed for which the shock terminating the local supersonic region passes the shoulder. This region extends with Mach number so that shock moves along the body, passing the stations at $3{\cdot}2D$ and $5{\cdot}6D$ at the higher Mach numbers shown in Fig. 3. Results for the free flight model (including a dummy sting) in Fig. 3, show the shock to move rapidly along the body to the tail at a little below $M=1$; the shock reaches the tail in the tunnel tests at progressively higher Mach numbers for the larger bodies and not until $M=1{\cdot}06$ for the largest model of 0·52 per cent blockage. These differences may not be of major significance for a cylindrical body from the point of view of effect on overall forces, for the latter depend more on the nose shape than on the length of the cylindrical portion of the body, but they could have important effects if control surfaces were affected.

Tunnel tests were also made with wings on the bodies; the results in Fig. 4 show that the shock (now the main wing rear shock) moves back rapidly to the trailing edge at a relatively low Mach number in a way not much affected by the diameter of the body. A comparison is made also with results for the 2·5 in. diameter model obtained in the 10 ft. × 7 ft. closed wall High Speed Tunnel, Farnborough; there is some difference between the results but possibly not much more than might arise from errors in blockage corrections for the closed wall tunnel. The tests in both tunnels were made without transition fixing on the wings; the Reynolds numbers based on model diameter were $0{\cdot}64\times10^6$ in the 3 ft. and $0{\cdot}25\times10^6$ in the 10 ft. × 7 ft. tunnels; a check made at $R=0{\cdot}25\times10^6$ on the same model in the 3 ft. tunnel showed no significant differences.

The wall interference on shock position for the ogive cylinder body appears to arise when the supersonic region following the shoulder expansion is large enough to be significantly influenced by the wall. Probably flow from the plenum chamber into the low pressure expansion region reduces the extent of the supersonic flow so that the terminating shock is farther upstream than in free air. The effect on the shock position appears less on the wing body combination, and this may be due to the more variable pressure field at the wall, with less systematic outflow or inflow effects. Although these effects can be minimised by reducing model blockage (not more than 0·4 per cent is used in

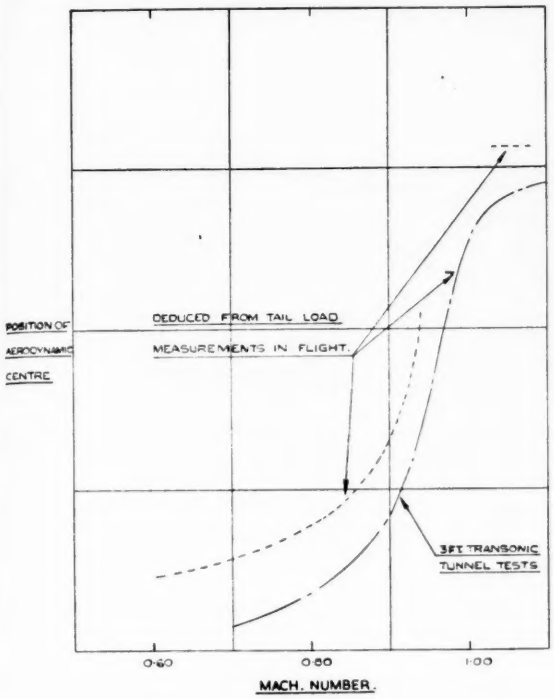

FIGURE 5. Variation of position of wing aerodynamic centre with Mach number.

3 ft. tunnel tests), development of the porous wall characteristics is desirable to reduce the outflow into expansion regions.

4. Results of Aircraft Model Tests

Some examples of results obtained in the 3 ft. tunnel transonic section will serve to illustrate the capabilities of these test facilities. In Fig. 5 is shown the variation with M of the position of aerodynamic centre of a model as found from wind tunnel tests; in the same figure are included results determined from flight measurements of tail loads. The agreement between two sets of results is reasonably good, with the tunnel values rather more than $0{\cdot}01C$ in front of the flight test values; this difference is about the maximum order of error already deemed acceptable.

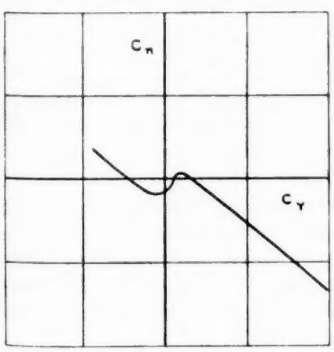

FIGURE 6. Yawing moment curve for basic aircraft at $M=1{\cdot}0$.

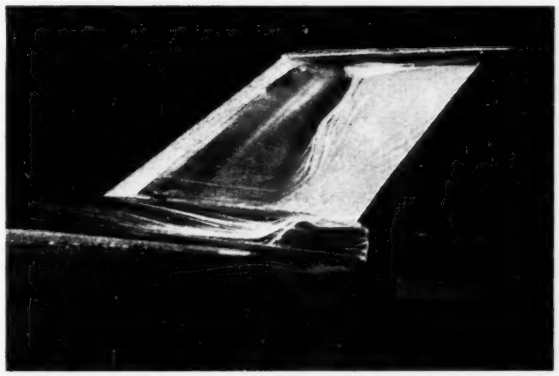

FIGURE 7. Oil flow picture for fin of basic aircraft at $M=1{\cdot}0$.

Interesting results have been obtained by correlating force and pressure measurements with flow observations, using oil with titanium oxide as a marking agent to indicate surface flow patterns. A special example of the use of this method occurred in the investigation of directional instability on an aircraft model[6]. The yawing moment curve in Fig. 6 shows an unstable slope at small angles of yaw. A reason for the instability is suggested by the surface oil flow on the model fin at zero yaw; a well developed line pattern can be seen in Fig. 7 over the forward part of the fin, but relatively unmoved oil over the rear half. This is interpreted as showing a strong shock across the fin with boundary layer separation behind it; the shock appears to originate from the wing trailing edge and falling across the fin is strengthened by the velocity field due to the fin and afterbody. A modification to the upper rear fuselage, which increased the cross sectional area and reduced the curvature, was then made with a view to reducing the shock strength by decreasing the velocity due to expansion at the fin and fuselage juncture. The results for low incidence given in Figs. 8 and 9 show the success of these modifications, the fin oil flow showing no separation and the force record no instability. Full scale qualitative handling tests of the afterbody modification confirmed the tunnel results.

5. Concluding Remarks

The significant effects on directional stability already noted, arising from relatively small changes in afterbody shape, are an indication of the need for caution in distorting fuselage shapes for the purpose of accommodating a sting and balance. In the foregoing example no distortion was necessary, but it is unavoidable for a fuselage tapering to a point. It appears particularly important to ensure that shocks associated with body distortions do not cross the wings or tail surfaces in a way likely to be unrepresentative of the full scale aircraft. In some cases extending the fuselage beyond the trailing edge may keep shocks away from the tail surfaces. It has been suggested that when increase in body diameter is necessary to accommodate a balance,

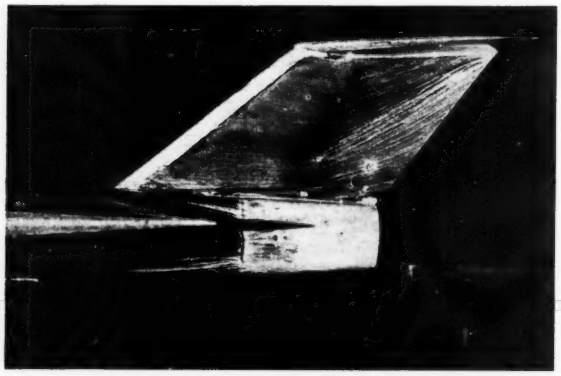

FIGURE 8. Oil flow picture for fin with modified afterbody at $M=1{\cdot}0$.

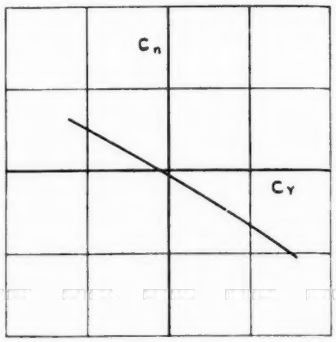

FIGURE 9. Yawing moment curve with modified afterbody at $M=1{\cdot}0$.

representative flow over the afterbody may be attained by keeping unchanged the profile shape of the rear axial area distribution.

In the testing of design projects, accommodating the balance without producing flow quite unrepresentative of full scale conditions is one of the main problems to be resolved. In research testing, a freer choice may be available, and theoretical studies likely to require wind tunnel investigations should, where possible, include a typical case with a body large enough to accommodate a balance, so that directly comparable results in transonic conditions are less difficult to achieve.

ACKNOWLEDGMENT

Some of the estimates of accuracy in Section 2.3 and of interference effects in Section 3 were made by Dr. L. C. Squire.

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Summary of the Discussions

by

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1. Introduction

At the start of the discussions, the Chairman (Major G. P. Bulman) gave a warning that parts of the subject of transonic wind tunnel testing were still classified and therefore he asked speakers to be careful in what they said. In spite of this, it was possible to discuss a wide range of topics without transgressing security restrictions. Those members of the audience who were familiar with the subject were able to put many questions to the lecturers and to raise several new points, while the large proportion of the audience to whom the subject was relatively new were able to gain a useful insight into the problems involved in the operation and design of transonic tunnels.

The audience numbered some 300 and 15 of these took part in the discussions which, together with the replies from the lecturers, lasted more than two hours. To keep this summary of the discussions to a reasonable length it has been necessary to abbreviate considerably the comments which were made.

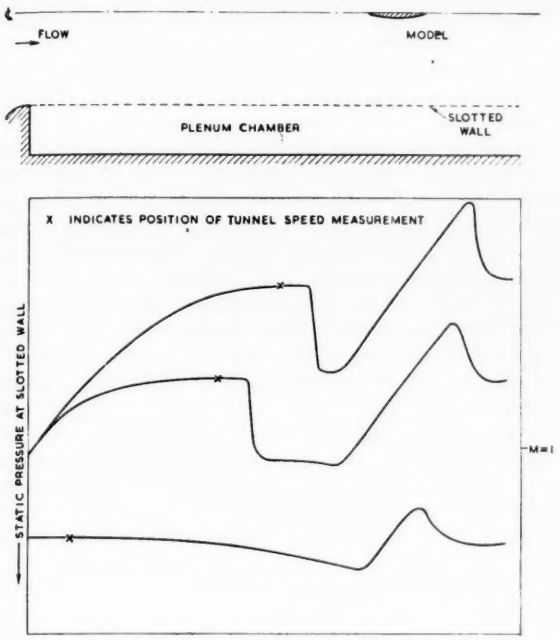

FIGURE 1.

2. Working Section and Tunnel Speed Measurement

A large amount of the discussion was spent on matters allied to the design of the working section. This subject was first raised by a speaker from N.P.L. who was worried by the problem of tunnel speed measurement. His major criticism of existing tunnels was that the length of working section ahead of the model was so short that the speed of the air stream could not settle down to a steady value corresponding to free stream conditions and, in addition, the speed of flow throughout the working section was likely to be influenced by the presence of the model. This point is illustrated in Fig. 1. At subsonic speeds the pressure at the wall will only settle down to a uniform value unaffected by the model at a point far ahead of the model. It would therefore be possible to measure the tunnel speed at a point far upstream. At supersonic speeds, on the other hand, if the supersonic flow is generated in a slotted working section, uniform supersonic flow is only reached some distance downstream of the beginning of the slots, the distance downstream increasing with increase in Mach number. The result is that the best position for tunnel speed measurement moves over a considerable distance as the Mach number is changed. The consequence of these considerations is that it could be difficult to determine the tunnel speed if the length of unaffected section ahead of the model is insufficiently long.

It had been suggested that tunnel speed could be measured from the plenum chamber pressure, but unless the length of working section were sufficiently long the plenum chamber pressure could be seriously affected by the presence of the model.

In addition to this, the length of working section downstream of the model should be reasonably long since, as was pointed out by another speaker from N.P.L., the pressure gradient along the wake behind the model provides an essential link between the free stream Mach number and the Mach number close to the model. The free stream Mach number in free air is reached at infinity downstream and to achieve similarity with free flight the tunnel flow must approximate to these conditions. The view of the speaker was that the length of working section downstream of the model is insufficient in many tunnels operating at the present time and that the extraneous pressure gradients caused by changes in tunnel geometry, or by the presence of tunnel supports, could have serious effects, particularly in experiments involving separation bubbles extending into the wake.

The lecturers agreed that the precise measurement of tunnel speed was a real difficulty and required further investigation. The almost universal practice at the present time was to use plenum chamber pressure for measurement of tunnel speed and one of the lecturers felt that the reason for the success which had been achieved using this method was the large size of the plenum chamber. He had made tests with a model of about half per cent blockage in which pressure measurements had been made at four or five points along the plenum chamber and these had not shown any significant variations. Other speakers felt that with a blockage of this order the effects were not serious, but if the model size were increased it was possible that serious effects could result. This could be checked by testing geometrically similar models of different sizes and it appears that such tests are planned.

It was suggested by another speaker that it should be possible to devise some direct method of measurement of tunnel speed rather than by pressure plotting. He suggested, for example, using sparks to provide hot spots in the air stream; the paths of these could be traced optically. Alternatively, the speed could be obtained by measuring the speed of sound propagated through the working section from a stationary source in the tunnel by the use of microphones. The lecturers felt that for research investigations, methods of measurement of this type would be valuable, but that considerable effort would be required to develop the required apparatus. For routine testing the use of plenum chamber pressure was generally sufficiently accurate and it was pointed out that even if the air speed could be measured directly, it was still necessary to find a point where the air flow is not affected by the presence of the model. A speaker from the Industry observed that the precise measurement of the tunnel speed was strictly of importance only in fundamental research. In applying the tunnel results to actual aircraft it should be remembered that in normal flight testing the Mach number could only be measured to within about 0·02.

Several speakers thought that the lengths of working sections at present in use might be inadequate, either to

permit accurate tunnel speed measurement or to prevent conditions downstream of the working section from influencing the air flow at the model. One speaker suggested that a length of three times the tunnel width or height was not enough and felt that the length should be at least four times the height. A further consideration was that the flat part of the Mach number distribution curve seemed to go a good deal farther back at the higher Mach numbers than one anticipated and, therefore, the highest speed to which tunnels could be run was sometimes restricted by the length of working section.

Questions were asked on the most suitable cross-section shape for the working section. The lecturers felt that this was a debatable question, but on the whole they favoured rectangular rather than square section, since they felt that with a square section there was a possibility of some form of focusing of disturbances in a similar way to that given by a circular section, although the effects would probably not be so severe.

3. Tunnel Drive

Reference was made by a speaker to a large industrial tunnel which employed a somewhat unusual method of varying the tunnel speed. The conventional method of speed control is by control of the speed of the motors and compressors. In the tunnel which this speaker described the control of tunnel speed is achieved by varying the mass flow through the compressor by means of a butterfly valve at the compressor inlet. Although this method of control is wasteful of power when running at low tunnel speeds, the initial cost is significantly lower than with a variable speed compressor drive. For such a drive it is generally necessary to use the Ward Leonard system and the associated machinery to deal with the power required by a transonic tunnel is very expensive. The speaker emphasised, however, that with this particular tunnel it was rarely found necessary to run at Mach numbers below 0·4 and consequently the amount of power wasted was not excessive. The running cost of this tunnel is of the order of £30 per hour and the working section is 22 in. × 20 in. To compare with this figure it was stated that the cost of the power for running a jet-driven tunnel of slightly larger size was of the order of £62 10s. per hour, although this would obviously depend on the speed at which the tunnel is run.

Another method of speed control which was mentioned was the changing of the blade angles of the fan. Opinions as to whether this was advantageous were divided; one of the lecturers felt that provided the swirl angle could be changed without stalling the blades, there was little to be gained from the ability to change the blade angles as well. He thought that by changing the swirl angle the entire range of speeds required in normal operation could be covered with acceptable efficiency. Another speaker felt that fixed blades were quite adequate for normal purposes.

Several questions were asked about the total power required to drive a transonic tunnel. Figures quoted covered a wide range and it appeared that most tunnels used of the order of 400-600 h.p. per square foot of working section. An exception was one tunnel which employed diffuser suction and in which the power used was of the order of 2,000 h.p. per square foot. It was suggested that the power required for auxiliary suction was normally of the order of half of the power of the main fans and, although this seemed a large quantity, it was pointed out that the pressure ratio required for auxiliary suction was much higher than for the main flow, being of the order of 3 : 1.*

Another speaker suggested that when designing a tunnel the important consideration was compressor pressure ratio, rather than power, since the pressure ratio fixed the Mach number at which the tunnel could be operated. Power could be looked after by changes in the density of the air in the working section. He suggested that when designing a tunnel it was important to keep the working section as large as possible and, therefore, under certain conditions it could be advantageous to run it at reduced densities. He felt that a tunnel should, in the first instance, be designed to run at reduced density; if additional power became available at a later date larger motors could be put on the compressor and the normal operating density of the tunnel increased.

4. Half Models and Model Supports

A member of the audience asked whether the lecturers considered half model testing to be valuable, or whether there were inherent difficulties in getting the right answers out of half model tests. He observed that half models were much easier and therefore quicker to make and that they gave the advantage of a greater Reynolds number for any given tunnel size. The lecturers felt that half model tests had their advantages, but problems of tunnel wall boundary layer and leakage made it difficult to obtain accurate results from them.

*In the A.R.A. tunnel the shaft h.p. required to achieve a Mach number of 1·2 at atmospheric stagnation pressure and 40°C stagnation temperature is 380 h.p. per sq. ft. of working section area. Approximately two-thirds of this power goes in the main fan drive and one-third into the auxiliary suction drive. This probably represents the minimum power for a tunnel of this type since the working section is relatively short, the tunnel circuit has a low loss and the perforated wall is of optimum design. No direct comparison with a transonic tunnel using diffuser suction is possible but analysis of various results suggests that the corresponding figure is about 460 to 480 h.p. per sq. ft., representing a 25 per cent increase in total power required. For smaller tunnels the power per sq. ft. will tend to increase, due to the lower Reynolds number, but this increase will only be of the order of 10 to 20 per cent. A much larger increase is possible if the open area of the perforated or slotted wall is changed or, with perforated walls, if the ratio of hole size to wall thickness is not optimum. For some combinations the tunnel power may be up to as much as 600 h.p. per sq. ft. The figure of 2,000 h.p. per sq. ft. mentioned in the discussion would appear to be so high primarily because the compressor used to drive this tunnel was working at a pressure ratio well below its design figure and therefore at a very low efficiency.—R. HILLS.

With low aspect ratio models it was felt that the parts of the wing affected by the boundary layer could be so large as to render any test results useless. One of the lecturers felt that the main use of half model tests was for comparative results at the highest possible Reynolds number for a given size of working section. Thus, half models could be used to find the effect of some modification to the wing plan form or the effect of external stores, provided that such modifications were sufficiently far outboard of the wing root.

A speaker suggested that the difficulties arising from the presence of the wall boundary layer could be got over by the use of vortex strips. These would remove the boundary layer from the wall in the vicinity of the model and worked very satisfactorily at low speeds. It was suggested, however, that while this technique was effective at low speeds it might prove to be difficult to use at speeds where comparatively small changes in the effective area could produce big changes in the velocity distribution. It was felt, however, that this idea was worth further investigation.

The necessity to use a long sting for model support could impose restrictions at the angle of incidence or the maximum loading at which the model could be tested, and a speaker asked whether difficulties had been encountered. Stings, of necessity, must be long as the support must be removed some distance from the model to avoid blockage of the working section. In reply it was stated that the restrictions were not severe for the majority of cases, but that the length of the sting could prove an embarrassment from the point of view of rig stressing. It was observed that it was taken for granted these days that wind tunnel models should be supported by a sting. In view of the importance of disturbances propagated upstream, it was suggested that for certain types of tests an alternative means of support was desirable. This was particularly true when it was necessary to distort the rear of the model to accommodate the sting. A speaker therefore wondered whether any attention had been given to the problem of holding the model by struts, such as had previously been used in high speed subsonic tunnels. In the subsonic tunnel they had gone out of favour because of their blockage effects, but with a slotted tunnel the blockage effects could be minimised. It was pointed out, however, that struts would give rise to shock waves and that these shock waves would have undesirable effects on the model, which could prove more unpleasant than the troubles arising from sting mounting. An alternative to struts, which was suggested, was to have a forward mounted sting supported from the contraction end of the working section. Although this would leave the tail end of the model clear it would suffer from the disadvantage that the boundary layer on the sting would pass over the model. This could cause particularly severe trouble if attempts were being made to reproduce the flow through fuselage intakes on the model.

5. Shock Wave Reflection and Blockage

A speaker asked whether in the A.R.A. tunnel it was possible at low supersonic speeds for the detached bow-wave of the model to lie in the nozzle, instead of the perforated section and, if so, whether any difficulties were experienced with either wave reflection or tunnel speed measurement. In reply it was stated that there was a possibility of the wave hitting the solid wall, but it was thought that it would only affect a small Mach number range. Another speaker said that in Mr. Kirk's paper it was implied, probably inadvertently, that the shock waves reflected from the tunnel wall on to the model were likely to be important only when they provoked boundary layer separation. Since, however, to cause a boundary layer to separate the deflection angle would have to be about 2° and the pressure ratio about 1·1 it was certain that shock reflection would be important long before the reflected shock wave was strong enough to provoke separation.

Several references were made to the problems arising when only two of the four walls were slotted or perforated. The perforations in such a case enable the tunnel to be run without running into serious blockage effects but, on the other hand, shock waves can be reflected from the solid walls. It was admitted that this was a great disadvantage. An advantage of using two solid walls was that it was possible to view the model and some of the shock waves using a schlieren system. It was, however, difficult to examine the reflections from the solid walls themselves because of the difficulties of looking through the slots.

Another speaker mentioned that he had made some tests on a body of revolution on which there was one longitudinal row of pressure holes. The model was tested in a tunnel with two slotted walls only and it had been found that there was no measurable variation in pressure distribution along the line of holes as the model was rotated. This indicated that plain walls had no significant effect in reflecting shocks on to the body in this particular case. It was said that if shock wave reflection effects were present they were much more easy to interpret and could therefore be more readily corrected than those effects caused by blockage.

On the question of wall venting a member of the audience observed that with plain walls there must inevitably be some conditions under which trouble with shock wave reflection could be experienced. He asked whether it was possible to eliminate blockage interference completely with two ventilated walls only and leave only the problems of shock wave reflection. The questioner was reassured that sufficient correction could be obtained for blockage in this way and furthermore, there were reports of a tunnel running successfully with only one vented wall.

It was pointed out that when considering the size of a model in a transonic tunnel the length could be more important than the blockage as, with a long model, there was a greater possibility of the reflection of shock waves from the nose of the model to the rear of the model.

An unusual aspect of reflection was referred to by a speaker from the N.P.L. who described flow past an aerofoil at high subsonic speeds. Expansion waves originating from the surface of the aerofoil was reflected

from the sonic line in the flow and if this sonic line ran into the tunnel wall then the flow at the surface of the aerofoil could be significantly affected. A result of this appeared to be that the shock wave could move an appreciable distance downstream, compared with the free flight condition.

Reference was made to tests by Kruger in one of the Swedish tunnels in which the drag of a body of revolution was measured as a similar body of revolution was gradually brought closer to it. These tests had shown that if the spacing between the two models was greater than about two model lengths, then the measured drag was about equal to the free flight drag coefficient of the body. This result might be useful as a guide to the influence of a slotted wall and suggested that the body should be greater than one model length away from the wall. The question of model size was extremely important in so far as it dictated the Reynolds number which could be achieved in the tunnel.

A speaker from the engine industry enquired whether it would be possible to make tests on intakes in a transonic tunnel using a larger blockage than the half per cent quoted for aircraft tests. He stated that as far as was known there were no discontinuities in the performance of intakes over a limited range of Mach number and, therefore, it would be reasonable to omit testing over part of the speed range and to interpolate for the results. He suggested that in these circumstances it might be possible to go up to a 5 per cent blockage. The lecturers replied that tests were being made on intakes with blockages of this order. These results were not yet available, but one thing which was apparent was that bow-wave position depended on blockage.

6. Diffusers and Contractions

Reference was made to flow instability in the working section resulting from upstream propagation of disturbances originating in the diffuser. The strength of such disturbances increases with the Mach number of the stream into which they are propagated as the Mach number is increased towards unity. Difficulties due to this have sometimes been experienced in slotted tunnels using diffuser suction, since the diffuser suction can give rise to flow instability in the diffuser. Flow instability in the diffuser had also sometimes been observed with jet-driven tunnels, but on the other hand others have been operated quite satisfactorily. A method of preventing the upstream propagation of such disturbances is to use a choke throat downstream of the working section. When such a choke is fitted it does, of course, control the speed in the working section.

A speaker asked on what basis diffusers were normally designed. He suggestec that diffusers could be expensive in power consumption and it would seem to be worth considerable effort to obtain the most efficient design. It was observed that model tests on diffusers were rarely satisfactory as diffusers were very sensitive to Reynolds number effects and, in addition, their performance varied significantly with the conditions of the boundary layer at the diffuser inlet and hence, on the tunnel design as a whole. The best basis for the design of a diffuser therefore seemed to be past experience and, in general, diffusers were designed on the basis of a cone angle of between $4\frac{1}{2}°$ and $5\frac{1}{2}°$.

Referring to diffuser suction a speaker from R.A.E. elaborated on the arrangements referred to by Mr. Vessey in his paper for the 2 ft. × 1 ft. tunnel. The problem was twofold, firstly to establish the necessary reduction in pressure to induce the flow from the plenum chamber, and secondly, to achieve efficient mixing between the two streams. To achieve the necessary suction it had been necessary to accelerate the main stream to a Mach number of about 2 at the flow re-entry position by use of a properly shaped effuser. Thus, in the diffuser it might be necessary to mix subsonic and supersonic flows and attention had to be given to the possibility of shock waves in the flow separating the boundary layer on the wall of the diffuser. Care must be taken in the design of the lip of the flap controlling flow of auxiliary air into the diffuser.

Careful design of the contraction was important in that it was possible for locally supersonic flow to be generated, which might give rise to disturbances within the working section. It was observed by a speaker that looking at some of the tunnels which were in existence one got the impression that the design of the contractions was quite arbitrary. He asked what methods were in general use for their design. One of the lecturers stated that the method which had been employed on his tunnel was to find a tunnel that had a satisfactory contraction and modify it very slightly to suit the particular requirements. A model test was then made which proved quite satisfactory and so the tunnel contraction was made to that shape. He admitted that he had no idea where the shape which was chosen actually originated. He stated, however, that the shape of the contraction near the throat was checked very carefully and using simple one-dimensional theory the Mach number gradient near the throat had been investigated. Other methods of contraction design which were mentioned were those by Harrop* and Batchelor.

In conclusion of the discussion on contractions it was stated that the N.P.L. were attempting to collect a library of the details of contraction design and it was requested that details of any particular contraction should be communicated to the N.P.L. where they would be available for general reference.

7. Oscillatory Tests

It was observed that most of the work which had been described in the papers referred to tests in steady flow. A speaker from the N.P.L. said that tests on oscillating models were important and it might be expected that in such tests tunnel interference effects might be rather different, on the grounds that there was varying flow into and out of the walls. This could be expected to lead to damping and hence to some sort of phase lag condition between the motion of the flow and the motion of the model. Some work had been done

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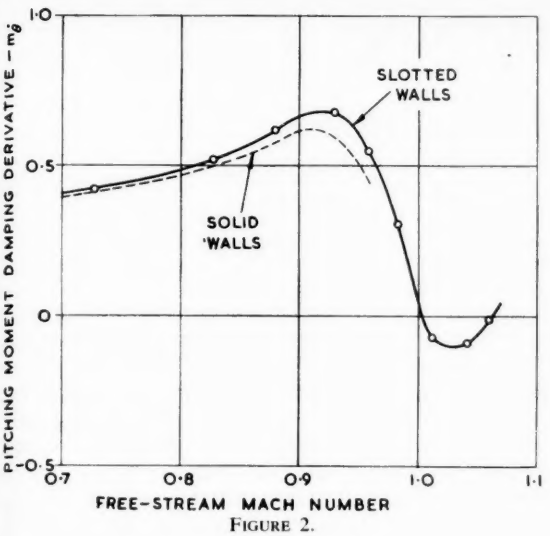

FIGURE 2.

on this subject at the N.P.L., but the main weakness was to find free flight tests with which to compare the tunnel results. The comparisons which had so far been made were between results measured in a slotted tunnel and corrected results made in a solid wall tunnel. These should reveal if there were any large damping effect. The test results are illustrated in Fig. 2 in which the pitching moment damping derivative for a half-span delta wing is plotted against Mach number for the two types of test. The results shown in this diagram are typical of the correlation which has been found in all the test results for highly swept or delta wings. For unswept wings and for two-dimensional wings, occasional anomalies have been found at the lower Mach numbers. Other speakers emphasised the importance of tests on oscillatory derivatives since a great many of the flight difficulties which had been encountered at transonic speeds were due to oscillatory, rather than static, derivatives. Tests which had been made in one industrial tunnel had given results which agreed very well with flight test results to a Mach number of about 1·03.

8. Conclusions

In the course of conversation with the lecturers following the meeting, the opinion was expressed that the discussion might have painted a blacker picture of the transonic tunnel than was actually the case. It was inevitable that a large proportion of the meeting should be taken up by discussion of the points in design and operation which caused difficulty, to the exclusion of those in which success had been achieved. This lack of balance in the discussion might easily create the wrong impression of the present status of transonic wind tunnel testing. It was felt that the meeting demonstrated that, while there were still many facets of tunnel design which required further research, the transonic tunnel had already proved itself a valuable tool in the investigation of problems arising in the design and operation of aircraft and missiles intended to operate at transonic speeds.

Flame Quenching

by

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This review of published theoretical and experimental information forms a background to the design and testing of flame-quenching elements.

1. Introduction

The possibility of explosions in a duct or chambers, which may contain a mixture of an explosive vapour and air, is a hazard commonly occurring in industrial plant, often necessitating a very strong and costly form of construction which would otherwise not be required. Where the structure is not strong enough to withstand explosion pressures serious damage results, frequently with injury to personnel.

If the explosion could be confined to a small volume of the total potentially explosive mixture, in the region where this was ignited, little or no damage would result. A device is required which will allow the almost unrestricted flow of gases under normal conditions and as a result of an explosion, without permitting the passage of either flame or gases at a high enough temperature to ignite the mixture beyond.

Such a device would be a flame-quenching element, or flame trap. It has been found that some kind of wall, consisting of many passages of small cross-sectional area, parallel to the direction of flow, is most effective in presenting a large area of cooling surface to the flame, without entailing a substantial pressure drop.

In formulating certain tests on flame-quenching, it was found necessary to study various aspects of combustion, including the means of ignition, ignition temperatures, velocity of propagation, experimental work on propagation in tubes and on limits of inflammability of various hydrocarbons in air and oxygen, as well as the theoretical work on flame propagation and quenching.

This paper deals with this study and is concerned with establishing a basis from which the mechanism of flame-quenching can be argued.

2. Definitions

To obviate any doubt as to the meaning of certain terms used in this paper, these are defined here. Some of these terms have, in the past, been applied with no precise meaning or with different meanings; "flame velocity," for example, has apparently been used to denote both burning velocity and velocity of propagation. It is much to be hoped that standardised terminology will be adopted by all in the field of combustion research.

Mixture: an inflammable gas, or gases, termed the fuel, together with oxygen, or air, or oxygen and some inert gas (*e.g.* nitrogen, argon or helium), the proportions of the constituents being the same throughout the volume.

Mixture strength: the ratio of the mass of fuel in the mixture to the total mass. It is, however, frequently expressed as a percentage ratio of volumes at the same conditions of temperature and pressure.

Stoichiometric mixture: that mixture which theoretically contains just sufficient oxygen to enable complete burning of all the fuel to take place.

Equivalence ratio: the ratio of the actual mixture strength to the strength of the stoichiometric mixture. Thus, for a stoichiometric mixture the equivalence-ratio is unity; for a weak mixture the ratio is less than unity; for rich mixtures the ratio is greater than unity.

Burning velocity: the volume of mixture burned per second per unit area of flame surface. The value is constant for a given fuel and for given conditions of temperature, pressure and mixture strength.

Velocity of (flame) propagation: the linear velocity at which a flame front travels along a given axis in the direction of propagation.

Detonation: flame propagation in which the flame velocity reaches the local velocity of sound in the gas. The propagation is therefore accompanied by a shock wave.

Cool flame: a species of flame which corresponds to only partial combustion of the mixture. It is of low temperature (less than about 450°C) and low luminosity; it can be followed by complete combustion of the mixture, the process then being referred to as two-stage ignition. The flame has a low velocity of propagation and is responsible for the low ignition temperature of the higher members of certain organic series. The property is shown by the following homologous series: monohydric alcohols, paraffins, and especially the ketones and ethers.

Ignition temperature: the lowest temperature of a mixture at atmospheric pressure (or other stated conditions) at which, in the absence of flames or sparks, any ignition of the mixture, including cool flames, can occur.

Limits of inflammability: the upper and lower limits of inflammability of a mixture are those mixture strengths at which continuous flame propagation will only just occur under prescribed conditions; between these limits some kind of propagation (not excepting

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cool flames) can always occur and outside them no continuous propagation can take place.

Heat sink: a body or part of a body which will absorb, with a negligible change in its own temperature, all heat energy which arrives at its surface.

3. Experimental Work

3.1. SIR HUMPHRY DAVY

Davy's work in 1816 was the first logical attempt to produce some device which could be described as a flame trap. For his miner's lamp the construction had to be such that the flow of air to the flame was not restricted but, if an explosive fire-damp atmosphere were present, the explosion which took place in the lamp would not cause ignition of the surrounding atmosphere.

In the course of his research he investigated the passage of flame through gauzes, through single tubes connecting vessels which contained explosive mixture, and through concentric circular canals. His observations, considered in the light of present-day knowledge, were extremely pertinent.

As is well known, the Davy lamp has layers of metal gauze as the safety device.

Until recent years no further experimental work of this nature was done and, even now, little fundamental knowledge has been gained; such flame traps as are in use were produced on a "try it and see" basis. The mechanism of extinction of flame is not understood, as is evidenced by the many theories.

3.2. VELOCITIES OF PROPAGATION IN TUBES

Workers in this country became interested in the measurement of velocities of propagation about forty years ago, largely in connection with research into coal-mine explosions. Velocities of propagation of methane-air mixtures were measured, usually by Wood's metal or "screen-wire" indicators. The screen-wire indicator consisted of one or more thin copper wires, stretched across the tube and heated electrically to red-heat; these melted as the flame passed. By mounting the indicators at several known positions along the tube, and recording the breaking of the circuit against a time base, the velocity of propagation could be estimated.

Searching for some standard of comparison, investigators recognised that the internal diameter and material of a tube would have some effect on the flame propagation. Usually the velocity of "uniform movement" was measured. It was thought that, for a flame started from the open end of a tube and travelling towards the closed end, the flame front after an initial period of acceleration travelled with appreciably constant velocity, called the uniform movement. Much of this work was supervised by Wheeler[87-95, 97]; Coward *et al.* also contributed[13, 15, 16].

TABLE II

Velocity of current of mixture (cm/sec)	*Velocity of propagation (cm/sec)*	*Difference*
0	287	287
23	582	559
43	952	909
75	1527	1452

Investigations about 1915 involved a repetition of those by Mallard and le Chatelier in 1883[59]; these results are reported in References (39) and (99).

Mallard and le Chatelier[59] stated that a tube diameter of 0·3 cm. was sufficiently large not to affect the velocity of propagation. However, Haward and Otagawa[39] showed that this supposition was incorrect by measuring the velocity in tubes of 0·9, 1·15 and 2·5 cm. diameter with Wood's metal indicators. For the different diameters of tube the differences in velocity were greater for weak mixtures than for rich. It was asserted that the influence of cooling by the walls of tubes became negligible when the thermal conductivity of the gaseous mixture was high, that is, with rich mixtures. These conclusions cannot be said to be established, as the method of measuring the velocity of propagation was not very accurate. Wheeler and Mason[88] state that flame indicators using a low melting-point metal are unreliable and that red-hot copper screen-wires are superior. Velocities were measured in tubes of diameters from 2·5 up to 97 cm. and it was concluded that in tubes of less than 5 cm. diameter the velocity of propagation was reduced by the cooling effect of the walls. Above 10 cm. the increase of velocity was then due to the effect of convection currents. Coward and Hartwell[16] obtained similar results with tubes of the same range of diameter; in addition they showed that the volume of gas burned per second per unit area of flame surface was constant for different tubes, having diameters in the ratio 1 : 2 : 4. This quantity has been termed the fundamental burning velocity. Linnett[58] summarises the methods available for determining the burning velocity and instances experimental work based on each method.

For velocity of propagation, representative results, quoted from Wheeler and Chapman[91], are given in Table I.

The figures in Table II, quoted from Wheeler and Mason[90], show how the velocity of propagation is affected by the velocity of the mixture before ignition:

It was concluded that the large increases in velocity of propagation were caused by turbulence in the mixture.

The effect of initial temperature on the flame velocity is reported by Wheeler and Mason[88]; the paper by Wheeler and Coward[99] gives a summary of all this work.

TABLE I

Uniform movement in tubes of different diameters. 10 per cent methane—90 per cent air mixture by volume

Tube diameter (cm)	0·45	0·56	0·72	0·81	0·90	2·5	5·0	9·0	30·5	96·5
Velocity of propagation (cm/sec)	33·5	41·2	46·3	47·4	48·0	66	92	104	168	250

(The values of velocity are apparently not as accurate as the number of significant figures implies.)

None of the workers who performed experiments with tubes of different materials gave convincing results to show that the nature of the material was of consequence. Parker and Rhead[68] measured velocities of propagation in tubes of lead, copper, iron and glass about 2·5 cm. diameter and 10 metres long. Thin strips of Wood's metal were used as flame indicators at stations one metre apart. It was claimed that the metal tubes, being better conductors than glass, cooled the burning gases, so reducing the velocity. It is unlikely that the cooling effect in a tube of this diameter would be appreciable and the differences were more likely due to differences in surface finish of the tubes, which were not of exactly the same diameter, and to errors in the measurements of flame velocity.

The effect of turbulence on the velocity of propagation is difficult to study and comparison of results almost impossible, since there is no way of measuring an absolute degree of turbulence. Turbulence may already be present in the flowing mixture or be generated by the travel of the flame itself. Gross[38] gives some information on flame-generated turbulence and Summerfield *et al.*[82] attempt to describe the physical structure of turbulent flames. Karlovitz[49] gives a turbulent flame theory derived from experiment. Olsen and Gayhart[67] state that velocity gradients of a high order actually quench the flame rather than enhance its propagation rate, thus indicating the existence of an optimum gradient for maximum flame spreading. Mickelsen and Ernstein[62] give similar results. While the possibility of quenching by the presence of high velocity gradients is admitted, the authors have shown that velocities of propagation are greatly increased by turbulence in streams flowing with mean velocities from 20 to 120 ft. per second. Velocities of propagation in butane-air mixtures of up to 800 ft. per second were measured in a 6 in. diameter pipe. No further published data can be found for high flow velocities.

With laminar flow there should be no increase in the burning velocity as the velocity of the stream of mixture is increased.

Given suitable conditions, "detonation" can be set up in a pipe. When this occurs the flame front travels at the speed of sound in the gas, which may be several thousand feet per second. There is a considerable pressure difference across the flame front, associated with a shock wave, and much damage can ensue. Usually the flame has to travel a distance of the order of 10 metres* before this state arises. The combustion of gas-vapour mixtures in pipes is dealt with by Henderson[43] and Smith[79], who give data on the explosion pressures achieved in industrial pipelines, and by Steinicke[80]. Steinicke used pipes 10–50 cm. in diameter and 40 metres long, with ignition at the closed end. Propane-air mixtures reached velocities of over 100 metres per second within a short space, and detonated at over 2,000 metres per second. Rapid expansion produced by the initial ignition sets the unignited vapour in motion and the resulting turbulence causes a large increase in velocity of propagation. A pressure wave builds up and detonation takes place. The distance travelled before detonation is about 10 metres, depending on the mixture strength, diameter of tube, and roughness of the walls. Earlier detonation results with rough walls, and Steinicke cites unidentified Russian experiments in which wire enmeshed in pipes led to detonation within one metre. He asserts that a "Davy" shield, consisting of layers of wire gauze, prevents detonation completely.

*This approximate figure applies to pipes of large diameter, say over 3 cm. Detonation can rarely be set up in pipes of smaller diameter, where the cooling effect of the walls becomes appreciable.

3.3. GENERAL STUDY OF FLAMES AND SPARK IGNITION OF GASES

Analysis of the plane combustion wave

The problem of surveying the distribution of important parameters in the reaction zone of a stable flame has been tackled both by thermocouple traverse and by optical methods: these methods are reviewed in detail by Linnett[58]. Klaukens and Wolfhard[50] made measurements in a "low-pressure" Bunsen flame, since at low pressures the flame is drawn out and enlarged and the size of the thermocouple element is not so great compared with the reaction zone. Ni-NiCr wires of 0·01 cm. diameter indicated almost the calculated temperature distribution across the flame, radiation being disregarded. Friedman[30] used wires of 0·0005 cm. diameter with a thermocouple bead of 0·0025 cm. diameter and corrected for radiation. Very lean flames only were studied. Burgoyne and Weinberg[9, 10, 85] using the distribution of refractive index, analyse the plane combustion wave produced by a flat-flame burner. Calculations give the distribution of temperature, gas velocity, acceleration, residence time and reaction velocities. The investigation deals with a lean ethylene-air flame.

Spark ignition and ionisation in explosions

The spark ignition of gaseous mixtures is dependent on the inflammation of a small spherical volume of gas, beyond a certain diameter of which a flame will be self-propagating. There will be some connection between this diameter and the diameter of a tube along which flame will just not propagate. The ignition of methane and some of the higher paraffins with air by capacitance and inductance sparks and by heated surfaces was studied by Wheeler *et al.*[47, 86] and calculations were made of the form of the temperature wave spreading from point and spherical sources. Measurements were made of the minimum ignition energy required and the curves were found to follow closely the reciprocal of the burning velocity for the range of mixture strengths. In attempting to establish, by the use of long and short duration sparks, whether ionisation or the purely thermal effect caused ignition, it was shown that the different energies required from capacitance and inductance sparks were not inconsistent with the effect being a purely thermal one. This view was supported by Coward and Meiter[14] with data on the minimum volumes of ignited gas from which propagation was established.

It has been thought that the degree of ionisation is in some way connected with the rate of flame travel, since propagation is claimed by some workers to be caused by the activation of the unburned mixture by ionised radicles produced in the flame front. Ionisation is also thought by some to be the governing factor in the effectiveness of an igniting source, whether it be an electric spark or a hot surface, but it is more likely to be only a thermal effect. Some qualitative information is given by Garner and Johnson[34] dealing with the effect of catalysts, by Garner and Saunders[33], and by Lind [56, 57].

Lewis and von Elbe[53] developed a theory of flame propagation from an instantaneous point source of ignition, showing that a segment of a flame stores a certain amount of excess energy, acquired at the expense of the energy content of the burned gas as the flame grows from a small sphere to its final size. At the origin the excess energy has to be furnished by the ignition source to the smallest flame capable of self-propagation; this can be visualised as a burning sphere of the smallest volume consistent with the condition that the rate of heat production should be equal to the rate of heat conduction to the surrounding unburned gas. The flame diameters calculated on this basis are compared with quenching distances. Lewis and von Elbe's experimental work[55] gives evidence of the flame-quenching effect of the electrode material, and data for the minimum energies of ignition and for quenching distances are given for mixtures at room temperature and at pressures from 0·2 to one atmosphere.

Friedman and Burke[31] criticise this theory on the score that energy transfer by diffusion of reactants and products is ignored; they assert that this energy transfer is comparable in magnitude with heat conduction. Semenov[75] had shown that, for a plane flame, the excess energy vanishes as the dimensionless ratio $(D\rho c/\lambda)$ approaches unity, where λ is the thermal conductivity, D is the diffusion coefficient, ρ is the density, and c the specific heat at constant pressure. For many gases the ratio $(D\rho c/\lambda)$ is equal to, or greater than, unity at the relevant temperatures so that there is no excess energy.

In comparing experimental quenching distances with minimum flame radii calculated by their theory from minimum spark energies for ten gas mixtures, Lewis and von Elbe obtain a correlation with 21 per cent average deviation and 54 per cent maximum deviation. But simply by assuming the minimum ignition energy to be proportional to the surface area of the minimum flame sphere, a better correlation was obtained by Friedman and Burke for the same ten mixtures. The average deviation was then 14 per cent with a maximum of 31 per cent.

Radiation

The subject of radiation in gaseous explosions is one about which there is little information. David[20] shows that, in coal gas-air and hydrogen-air explosions, the heat lost by radiation during the explosion period is of the order of 5 per cent and during subsequent cooling about 10-20 per cent of the total heat liberated. Some information about the emissivity of flames is given by Ribaud[72]. It can be argued that, in the heat transfer from a flame entering a tube, the process of loss of heat would not take place quickly enough by conduction and convection, and the only heat transferred to the passage walls would be by radiation. The effectiveness of a surface would then depend on the total emissivity principally and on the diffusivity $(k/c\rho)$* as a measure of the capability of the first layers of the solid in keeping down the wall temperature.

TABLE III

Material	*Total emissivity (approx.)*	*Diffusivity*
Glass—ordinary	0·94	0·0057
Aluminium	0·05	0·826
Brass, bright	0·07–0·1	0·339
dull	0·25	
Copper, polished	0·07	1·133
oxidised	0·70	

From Table III it would appear that metal tubes would be many times more effective than glass in quenching a flame. It has, however, been shown by many workers (*see* later) that similar metal and glass tubes quench as effectively. It follows that processes other than radiation are taking place.

The radiation of flames is dealt with in books by Gaydon and Wolfhard[35] and by Thring[84].

3.4. THE LIMITS OF INFLAMMABILITY OF GASES AND VAPOURS

The methods of determining the limits of inflammability have fallen into three classes: (*i*) the criterion of continued propagation along the length of a tube, the diameter of which was sufficiently large to allow wall effects to be neglected, (*ii*) ignition of the whole volume of an explosion chamber, and (*iii*) extinction limits in a burner. Provided suitable precautions are taken the limits of inflammability at atmospheric temperature and pressure should be almost the same in each case. Coward and Jones[18] summarise all the information published up to 1952 on the limits of inflammability of 155 gases and vapours; they give 365 references. Egerton[23] gives a survey of the methods of determining limits of inflammability.

No satisfactory explanation for the existence of weak and rich limits has been put forward, although the effect on these limits of various additives and changes in physical properties of the mixtures has been studied, for example by White[100, 101, 102]. For both weak and rich limits the temperature of the flame is very much less than that for the stoichiometric mixture. Consequently, at the limit, flame is not self-propagating and, although burning can be started by an igniting source of sufficient strength, it dies out within a short distance.

The range of inflammability is widened at high temperatures and pressures, and narrowed at low temperatures and pressures. The effect of turbulence on the limits is uncertain, although there is some

*Where k is the thermal conductivity, c the specific heat, and ρ the density of the material.

evidence that the range is narrowed as the velocity of the mixture is increased.

3.5. IGNITION TEMPERATURES

The minimum spontaneous ignition temperature with air at atmospheric pressure is the lowest temperature at which any ignition of the mixture, including cool flames, can occur. One method of determining the ignition temperature is to heat a closed vessel to the test temperature, evacuate the vessel and admit the prepared mixture. The process is repeated with successively higher temperatures to establish the correlation of pressure, temperature and time delay. In one flow method the gases are passed through concentric tubes and mixed at the ignition temperature. In the "drop method", which is employed for inflammable substances normally liquid at atmospheric conditions, a small crucible is heated to the test temperature and a drop of the liquid is allowed to form in a small hole in the lid. Ten to 15 seconds are allowed for ignition to occur.

The table below gives some ignition temperatures in degrees Centigrade:—

Methane	650	Iso-Octane	530
Ethane	530	Ethylene	485
Propane	500	Acetylene	335
n-Butane	480	Benzene	500
n-Pentane	470	Ethyl alcohol	440
n-Hexane	275*	Acetone	440
n-Heptane	255*	Di-ethyl ether	180*
n-Octane	245*	Hydrogen	520

* NOTE: Cool flames.

It is seen that the paraffins have an ignition temperature of about 500 degrees Centigrade.

Information on ignition temperatures is given by several workers: Coward and Dixon[17], Frank and Blackham[27], Mullins[65], and Scott *et al.*[74].

3.6. QUENCHING BY WALLS AND SOLID SURFACES

The quenching of flame by proximity to solid surfaces may be due to a purely thermal effect involving the conduction of heat to the surface, thereby cooling the flame gases; the surface may absorb heat by radiation; or there may be some chemical effect which destroys chain carriers and so arrests the progress of the reaction.

Wheeler and Grice[96], in discussing perforated plate protection for electrical apparatus (plates of thickness from 0·125 to 0·5 in. and holes from 0·06 to 0·09 in. diameter), considered that the quenching effect was due both to the conduction of heat to the material surrounding the orifices and, to cooling by the rapid expansion of the gases as they passed from the region of high pressure within the vessel to the region of atmospheric pressure outside.

Jost[48] mentions the cooling effect of the walls on the progress of flame along a tube and shows theoretically that any heating of the surface penetrates only a layer of very small thickness.

Friedman[29, 32] measured quenching distances between parallel plates for propane-air flames at pressures from 0·08 to 2·8 atmospheres, and air-propane ratios (by weight) from 11 to 24. Other tests were made at atmospheric pressure: (*a*) with the temperature of the quenching plates varied from 80°F to 715°F, (*b*) with the temperatures both of the inlet gas and the quenching plates varied from 80 to 545°F, and (*c*) with a series of six different solid surfaces. No measurable differences in the quenching behaviour were observed for either lean or rich mixtures, or for any of the following surfaces: nickel, platinum, bright gold, roughened gold, potassium chloride, and water glass. The result indicates that neither the catalytic power of the surface nor the absorptivity for radiation significantly affects the quenching phenomenon.

Simon *et al.*[78] interpret the quenching effect in tubes on the basis of the destruction by cold surfaces of the chain carriers from the oxidation reaction. They correlate experimental data with equations for velocity of propagation and for values of critical diameter based on a theory of diffusion of active radicles. In such theories of flame propagation the reaction transfer is assumed to be by "active particles" or "active radicles" (*e.g.* O, OH and H, the oxygen, hydroxyl, and hydrogen active particles respectively), rather than by conduction of heat from burning to unburned gas. For this process, the nature of the walls would be important. Elston[25], however, has shown that the coating of surfaces with salts of various chain-breaking efficiencies has no measurable effect. Thus it seems unlikely that the destruction of chain carriers by the walls is important in quenching. The presence of a cold surface would lower the mean temperature of the burning gas in tubes and therefore the concentration of chain carriers. Hence the lowering of mean temperature and the destruction of chain carriers are hardly two rival explanations of quenching.

3.7. EXPLOSION-PROOF ELECTRICAL APPARATUS

Much research has been done by the Safety in Mines Research Board and the U.S. Bureau of Mines on methods of making safe the electrical equipment for mines. In many mines the presence of sufficient fire-damp in the atmosphere to form an explosive mixture is a frequent occurrence. This mixture, entering the casing of an electric motor or switchgear, could be ignited by sparks, and the burning gas cause ignition of the fire-damp throughout the mine. Thus, gases hot enough to ignite the mixture must be contained within the electrical gear and, to do this, the casings must either be strong enough to withstand the maximum explosion pressure or be vented in such a way that issuing gas is cooled below the ignition temperature. The first method is usually uneconomical and, for the latter, the flange joints are spaced to allow gas to escape.

Some of the earliest experiments on containing the high-temperature flame but allowing the escape of gas were performed by Beyling[7]. Wheeler and Statham[98] summarise the work of the Safety of Mines Research Board up to 1930. Rainford and Statham[71] also report on this work and Rainford[70] gives a review of flame-proof testing between 1922 and 1947. The design adopted was of flanges with a minimum width of one inch and a gap of about 0·020 in. James[46] showed

that flame passes more easily when the flanges are at an elevated temperature.

3.8. FLAME SUPPRESSION BY SINGLE TUBES

Wheeler and Payman[93, 94] determined lengths of brass tube required to prevent the passage of methane-air and methane-hydrogen-air mixtures, concluding that there was a limiting diameter of about 0·3 cm.; that is, whatever the length of brass tube, no flame would pass through it if its diameter were less than 0·3 cm. In transmitting flame from one chamber to another through a given small tube the size of the explosion vessel did not affect the results. It was found that the limiting diameter for the propagation of hydrogen-air mixtures was very much lower than that for methane-air mixtures.

Helmore[40] investigated the passage of flame along a tube between two explosion vessels and gave results for the length of glass and German silver tubes required for the suppression of explosions in various hydrocarbon-air mixtures. The dimensions of the cooling surface employed depended on the gas, mixture strength and physical state, whether turbulent or stagnant. Secondary factors included the explosion pressure and material of the cooling surface. His work was preliminary to the design of a successful induction pipe flame-trap for internal-combustion engines. Gauze, being unable to conduct away a sufficient quantity of heat, would not provide for the contingency of mixture burning at the trap and, instead, tubes were used as a cooling medium. Inconsistent results were at first obtained because of a varying degree of dryness of the tube walls; with a damp surface the power of extinction was the same for metals with a range of conductivity from 0·070 (German silver) to 0·918 (copper). Misleading results were also obtained when water was deposited from the products of combustion. Later, the apparatus was dried by heating and evacuating and the incoming gases were dried over calcium chloride.

As a result of these precautions Helmore claimed it was possible to distinguish between results obtained with different surface conductivities and small differences of length and diameter, but did not substantiate this claim by quoting experimental results. The thickness of the tube walls had no appreciable effect; the quantity of heat available, he said, would be taken up by the thinnest of tube walls obtainable (0·02 cm.).

He concluded that the internal diameter of the tube was the most important factor in preventing the passage of flame. Owing to the low heat-conducting properties of the gas, the central zone burned out of contact with the walls, and was insulated from them by products of combustion or unburned gas. The smaller the diameter the less the distance between the inner core of gas and the conductive walls of the tube. The effect of length was explained as follows. If the rate of extraction of heat by conduction and radiation exceeded the rate of generation of heat by burning, the length of tube required to extinguish the flame depended directly on the rate of propagation and rate of extraction of heat. Extinction would then be inevitable and the distance travelled would depend on the time required to reduce the flame temperature below that of spontaneous ignition.

Holm[44, 45] amassed data on the limiting hole and tube sizes for propagation of methane-air flames. The figures he gives relate to critical diameters for holes in copper foil, thin mica sheets and for metal and glass tubes, and appear to show that the hole diameter is the only important factor, there being very little difference between the values for sheets and tubes. The figures for mica and copper sheets are almost the same.

Further data are given by Berl and Barth[2] and Müller-Hillebrand[64].

With regard to flame travel in tubes, it is suggested that, for given conditions, there is a diameter below which flame will not propagate whatever the length of the tube. In a range of diameters greater than this critical value, but low enough for the thickness of the non-ignited boundary layer at the flame-front to be comparable with the tube diameter, propagation can continue until extinction takes place through heat exchange to the walls. There is thus, for diameters in this range, a critical length which will suffice for extinction. Above a certain diameter it seems likely that cooling to the walls will be negligible, so that propagation will continue indefinitely whatever the diameter of the tube. In all cases it is assumed that the tube is initially full of quiescent homogeneous mixture.

3.9. FLAME TRAPS

The history of the development of flame-arresting devices begins with the piece of research by Sir Humphry Davy[21] when set the problem of producing a miner's lamp completely safe even in an explosive atmosphere of fire-damp. Davy's reports are detailed and many of his observations very relevant to the wire gauze and passage types of trap. His concentric circular "canals" were very similar to Helmore's design of flame trap (described later), except for the spacing corrugations.

Sir Humphry's "chemical fire sieves" separating flame from air were safe canals, tubes and wire gauze tested thoroughly as the medium of communication between a large glass vessel full of the strongest explosive mixture of hydrogen and air, and a bladder, full of the same mixture. "The mixture in the glass jar was sparked and lambent flame played around the mouths of the safety apertures, but the mixture in the bladder did not explode."

Davy suggested that the burning gases are "cooled below the explosive point by passing through wires even red-hot." He showed that wires kept at red heat by an electric current did not ignite an inflammable mixture of methane and air. "In comparing the power of tubes of metal and glass, it appeared that the flame passed through glass tubes of the same diameter; the phenomenon probably depends on the heat lost during the explosion in contact with so great a cooling surface, which brings the temperature of the first portions exploded below that for firing the other portions. Metal is a better conductor of heat than glass; and it has already been shown that fire-damp requires a very strong heat for its inflammation."

Helmore *et al.*[41, 42] began with tests which included the examination of various factors governing flame extinction by surface cooling. In developing the induction pipe flame trap for spark-ignition engines the following results were obtained. The conductivity of the material was relatively unimportant, gauzes were not reliably effective and their resistance to flow was too great, and the wall of packed tubes became the design of trap adopted. As the length of channel decreased so the cross-sectional area of the passages had to be decreased to secure extinction. A satisfactory flame trap was developed using alternate plain and corrugated strips of German silver 0·002 in. thick wound in spiral form. The final design had triangular passages about 0·030 in. high and one inch long and was effective with all gasoline-air mixtures in an induction manifold. An expansion in the manifold to reduce the velocity through the trap itself was recommended. The pressure drop across the trap was measured and an expression formulated to relate the pressure drop to the undisturbed velocity of flow.

In their analysis, however, Helmore *et al.*, failed to consider the length of the flame trap passages and stated that, for small pressure differences, the resistance to flow P, expressed as a pressure drop, is a function of m the hydraulic mean depth (*i.e.* the ratio of cross-sectional area to perimeter), μ the viscosity, ρ the density, and V the undisturbed velocity of flow. The expression formulated for the pressure drop, although it contains m, will not predict P for a trap with a slightly different value of m. To obtain a general expression the length and the mean velocity in the flame trap passages themselves must be considered. It is unlikely that any expression would be satisfactory, for the tests were in the critical range of Reynolds number (300 to 3,000).

Discs made of sintered metals have also been tested as flame traps. Small discs were successfully employed by Egerton *et al.*,[24] and by Leah at Leeds University (quoted in Reference 24). A large sintered metal trap would not, however, be practicable owing to the material's weakness and to the large pressure drop which might be expected. In assessing the usefulness of a means of suppressing flame propagation the resistance to gas flow and the pressure drop which this entails must be taken into account. Annand[1] gives information on the resistance to air flow of wire gauzes; porous metal compacts are dealt with in Egerton's paper and by Taylor[83] and Grootenhuis[37], the latter giving information for discs of various diameters and particle size. The information leads to the conclusion that neither of these media is suitable for dealing with high rates of flow.

Radier[69] describes a typical industrial flame arrester, combined with explosion relief, while Steinicke[80] mentions the use of gauze shields, in pipes 40 metres long and 10–50 cm. diameter, to prevent detonation.

The occasional occurrence of serious explosions in the crankcases of large diesel engines, such as are used for ship propulsion, has led to recent investigations. The reasons for these explosions are discussed by Lamb[51], Montenero[63] and Burgoyne and Newitt[12]. Freeston *et al.*,[28] discuss the causes of explosions of this kind, and describe warning devices which may be used. The use of a flame trap-cum-venting device is mentioned, and the results show that to provide complete protection it is necessary to prevent the flame from spreading throughout the entire crankcase. Mansfield[60] gives results of tests on a crankcase explosion relief-valve 6 in. in diameter. Gauzes were not reliable when subjected to repeated explosions, but it was discovered that oil wetted gauzes were much more effective, both for the normal relief-valves and for partition flame traps separating the crank chambers of a multi-cylinder engine. Whether the oil merely made the holes in the gauzes effectively smaller, or evaporation of the liquid was instrumental in keeping the gauze cool, is not certain, but the use of wetted flame traps shows promise.

It has not been established conclusively whether the quenching effect of a small passage is due only to its geometry or, whether the nature of the walls is of importance. It should be possible to determine whether different material surfaces have any major effect on the efficiency. The use of tubes, similar except for their diffusivity, should help to establish whether the efficiency as a heat sink is dependent on this property or whether the cooling action on a flame is so rapid that any solid surface is equally effective, especially when, as is usual, the immediate surface is contaminated with dirt, oxide film and water vapour.

4. General Theories of Flame Propagation and Quenching

4.1. THEORIES OF FLAME PROPAGATION

It would be difficult to describe all the theories of flame propagation which receive consideration in present-day combustion work. Such is the state of knowledge that practical study of flames goes ahead with only general formulating equations for flame behaviour, certainly with no general theory to predict such phenomena as velocities of propagation, temperatures and limits of inflammability. Some theories do show reasonably good correlation with practical results for velocities of propagation and temperatures, but cannot be said to be convincing in view of the assumptions involved.

Marjorie Evans reviews the theoretical concepts of steady-state flame propagation up to 1952[26]. This work details the more important theories and a copious set of references is included.

Flame theories are nearly always formulated so as to predict the velocity of flame propagation of a given mixture and, as it is usually necessary to consider the progress along a tube, heat transfer to the walls and hence the possibility of quenching has to be taken into account. The application of many theories is limited by lack of data on the physical properties of the flame gases such as temperature, thermal conductivity, reaction rate, specific heat and density.*

*Some data on the physical properties of gases at high temperatures are, however, given by ten Bosch[8], Gregory and Archer[36], Nutall[66], Ribaud[73], and Stops[81].

Daniell's theory[19], representative of a number of others, was based on thermal considerations and developed from an earlier theory of Mallard and le Chatelier[59]; an endless tube was considered, the velocities of propagation assumed to be low compared with that of sound, and the pressure variations negligible. By the choice of a suitable coefficient an allowance was made for cooling. The value of this coefficient was dependent on the tube radius, and Daniell stated that there would be a minimum radius and velocity below which propagation would be impossible.

Another thermal theory by Semenov[75] leads to expressions for burning velocity in terms of temperatures, heat of combustion, mixture strength, conductivity and energy of activation.

Recent theories have been based on the concept of diffusion of active radicles, taking into account the rates of diffusion of hydrogen, oxygen and hydroxyl radicles. The rates of diffusion and concentration of these radicles at any instant are directly dependent on the temperature. The equations derived from one assumption for the mode of propagation are therefore directly comparable with those from the other and any agreement with experimental results does not justify the adoption of one theory only, either thermal or based on diffusion.

4.2. THEORIES OF QUENCHING

Lewis and von Elbe[54] discuss a theory of ignition, quenching and stabilisation of flames of non-turbulent gas mixtures. They describe the disturbance of a combustion wave by a "sink," such as a solid surface. This may be a heat sink; or it may be a chain carrier sink, at which all chain carriers or active radicles are rendered inactive. A form for the temperature distribution in a flame travelling within a tube is assumed, and this allows the calculation of limiting diameters for flash-back and quenching. The figures are tabulated for comparison of calculated and experimental values.

Friedman[29] states that the measurement of quenching distance provides an experimental approach to the study of flame structures. The mechanism by which quenching occurs is not well understood even qualitatively. Before the importance of free radicles and atoms in flames was fully appreciated, it was accepted that quenching was entirely governed by heat conduction from the flame to the cold surface, the chemical reaction being able to proceed only in those regions of the gas where the temperature was greater than a vaguely defined ignition temperature. In the light of present knowledge of chain reactions, however, it appears possible that diffusion of chain carriers to the wall may play a significant role in quenching. The later results of Friedman and Johnston[32] show that the quenching effect is independent of the nature of the surface for nickel, platinum, bright gold, roughened gold, potassium chloride and water glass. This result indicates that neither the catalytic power of the surface nor absorptivity for radiation significantly affects the quenching phenomenon. This result has also been demonstrated by Lewis and von Elbe[55, pp. 61-68] for glass surfaces coated with salts of high chain-breaking efficiency.

In order to apply the diffusion model of flame propagation to the process of flame quenching, it is assumed that the most important effect of a surface near a propagating flame is the destruction of chain carriers on the surface. Then, when a propagating flame approaches the entrance of a narrow duct, the first effect of the surface occurs in some thin cross-section ahead of the flame-front. If the number of molecules that react in this thin cross-section is sufficient for combustion the flame will pass into the duct; otherwise it will be extinguished. The flame-front is considered as a region of high temperature which is the source of the thermal equilibrium number of chain carriers—*i.e.* hydrogen and oxygen atoms and hydroxyl radicles. The chain carriers diffuse ahead of the burning zone, initiate the oxidation reaction, and are in turn regenerated in the chain reaction. Chain branching is negligible and chain carriers are destroyed by collision with the surface. The extent of the oxidation reaction is limited by the destruction of chain carriers.

The agreement between experimental data on quenching and a theory of quenching based on diffusion of chain carriers has been cited as evidence that the latter mechanism (rather than heat conduction) is of decisive importance in flame propagation. However, the equations governing flame propagation by diffusion of chain carriers and by heat conduction have been shown by Semenov[75] and by Friedman and Burke[31] to be of the same form.

The ignoring of heat conduction is particularly difficult to justify in view of the fact that heat flow of considerable magnitude is known to take place from a flame to a nearby cold wall, while the significance of free radicle diffusion under such circumstances is purely conjectural.

Such experimental evidence as exists, while far from conclusive, indicates that quenching is not controlled by such diffusion. The most compelling evidence against the preponderant effect of chain carrier destruction is provided by Lewis and von Elbe[55]; they showed that limits of inflammability were the same in tubes with widely different surfaces (clean Pyrex, and Pyrex coated with sodium tungstate, potassium chloride, or silver). These surfaces cover a very wide range of efficiencies for breaking chain-reactions.

Simon[76] compares experimental values of the low pressure limits for ignition with those calculated (as a function of tube diameter) from a diffusion theory and states that the correlation strongly suggests that it is wall-quenching which limits flame propagation in a tube. Later, however, she states[77] that diffusion effects do not usually give unique explanations of experimental data and that purely thermal explanations may be just as valid.

The work by Berlad and Potter[4, 5, 6] and Berlad and Belles[3] may throw some light on flame-quenching theory. The first report deals with the effect of channel geometry on the quenching of laminar flames. This effect is compared on the basis of average active-particle chain lengths for six different surface geometries: plane

parallel plates of infinite extent, cylindrical tubes, rectangular slots, cylindrical annuli, and tubes in the shape of ellipses and equilateral triangles.

The quenching of propane-air flames by a series of rectangular slots, cylindrical annuli, and cylindrical tubes was investigated over a range of equivalence-ratios from 0·82 to 1·30 and a range of pressures from 0·08 to 1·0 atmosphere.

A tube having a centrebody of small diameter provides much better quenching than a plain tube. This is explained by Berlad and Potter on the grounds that the small centrebody changes the distribution of active radicles. For a cylinder the distribution has a maximum at the centre, whereas for an annulus the concentration is zero throughout the centrebody. The average concentration of active radicles in the channel is thus said to be lowered, so greatly improving the quenching. A similar argument would apply to the distribution of reaction temperature.

Berlad and Potter state that, although the relative characteristics of the different geometries are here derived from the quenching equations based on the diffusion theory developed by Simon *et al*[78], they can also be derived from equations based on purely thermal assumptions. The conditions for flame propagation are then determined by some critical value of the reaction temperature. Thus, whichever assumption is made, a suitable set of equations will allow the prediction of the effectiveness of any of a large number of surface geometries, once the effect for any single geometry has been determined.

Moreover, Berlad and Potter[5] point out that the use of the diffusional equation given by Simon *et al.* is limited to stoichiometric or weak mixtures; by converting to a thermal analogue equation they extend its useful range to include rich mixtures. Two equations are tested, one based on the reaction between active radicles and the fuel, the other based on that between oxygen and the fuel: it is shown that, if the oxygen-fuel reaction is the one which controls the rate of burning, then the equation based on this reaction satisfactorily fits experimental data for the quenching of both rich and weak propane-oxygen-nitrogen flames.

The variation of the quenching distance with pressure is also discussed. It is shown that the pressure at which quenching just takes place with fixed separation of parallel plates has a variation approximately parabolic with mixture strength, having a minimum for stoichiometric mixtures. The pressure-dependence of the quenching distance does not vary appreciably with the amount of nitrogen contained in the mixture.

5. Bibliography

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Supersonic Jet Deflection

by

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1. Introduction

Appreciable forces can be obtained for steering a vehicle by the use of aerofoils whenever the relative air velocity is high and the air density is not too small. In practice this means velocities greater than 100 ft./sec. and altitudes below about 50,000 ft. Even when they are not in use for steering, aerofoils necessarily add to the drag of the vehicle.

All jet propelled vehicles can be steered by controlling the line of thrust of the jet, but this method or "jet control" is only effective when the motor is in action. It is possible, however, to obtain considerable control force without any increase in weight of the auxiliary equipment and, also, to obtain the control without any drag losses. The available steering force is also independent of both altitude and the relative air velocity. It can thus be particularly effective during the period of launch and at high altitudes. Jet control systems, compared with moving aerofoils at high altitude, have a lower weight and, in some cases, only small operating forces are required; thus they can be designed to give very rapid response.

The major disadvantage of jet control systems is that no steering control is possible with the motor cut out. Although the theoretical maximum side thrust can be the full axial thrust of the motor, the practical maximum is probably only 50 per cent of this, and in some control systems only 20 per cent. Most jet deflection methods require moving parts to be in association with high temperature gases.

This paper describes experiments which were carried out with model nozzles to examine some of the possible means of effecting jet control and to provide some data concerning the forces involved.

NOTATION

A area (wetted area of vane) (sq. in.)
C_D coefficient of drag
C_L coefficient of lift
D drag (=loss in axial thrust, ΔF_D) (lb.)
F_D axial thrust (lb.)
F_R resultant thrust (lb.)
F_s side thrust (≡lift) (lb.)
$F_{s\,(0)}$ side thrust at zero vane incidence (lb.)
h height of vane above nozzle exit plane (in.)
K constant
L lift (lb.)
l vane chord (in.)
M_c control moment (lb.ft.)
M_e Mach number at nozzle exit plane
M_∞ Mach number in the undisturbed jet
p_0 reservoir (stagnation) pressure (lb./in.²)
q, r vane position parameters (in.) (*see* Fig. 4)
R jet radius (in.)
s tube length (in.)
t vane thickness (in.)
u velocity (ft./sec.)
u_e jet velocity at the nozzle exit plane (ft./sec.)
w vane span (in.)
x length of bevelled extension (in.)
y half wetted span of vane (*see* Fig. 12) (in.)
α, β vane axis positions (*see* Fig. 4)
ε shock angle
ρ density (lb./ft.³ *g*.)
θ angle of inclination of a deflector
ϕ resultant angle of jet reaction.

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2. Methods of Jet Control

There are four ways of changing the direction of thrust of a jet motor:

1. The axis of the complete motor, or the combustion chamber and nozzle, or the nozzle alone, may be orientated in the required direction. This will be called "nozzle alignment."
2. The motor may incorporate small auxiliary nozzles on axes perpendicular to, or at an angle to, that of the main nozzle. This will be called control by "auxiliary side jets."
3. The main jet may be deflected in the required direction by applying a force. This is deflection by recompression and will be called "compression deflection."
4. The nozzle may be distorted so as to accelerate the gas asymmetrically, or the jet boundary may be disturbed and so produce a jet with a line of thrust other than the geometrical axis of the nozzle. This is deflection by asymmetrical expansion and will be called "expansion deflection."

2.1. NOZZLE ALIGNMENT

The thrust magnitude and direction are at once known in terms of the motor performance and the control operating mechanism. The steering power requirement may be very small because there is only bearing friction and motor inertia to be overcome. The magnitude of the side thrust is independent of the control force applied.

The design of a nozzle alignment unit however, can present many practical problems. Aligning the whole rocket motor is only feasible when it is a small part of the complete vehicle. Thus the method is usually

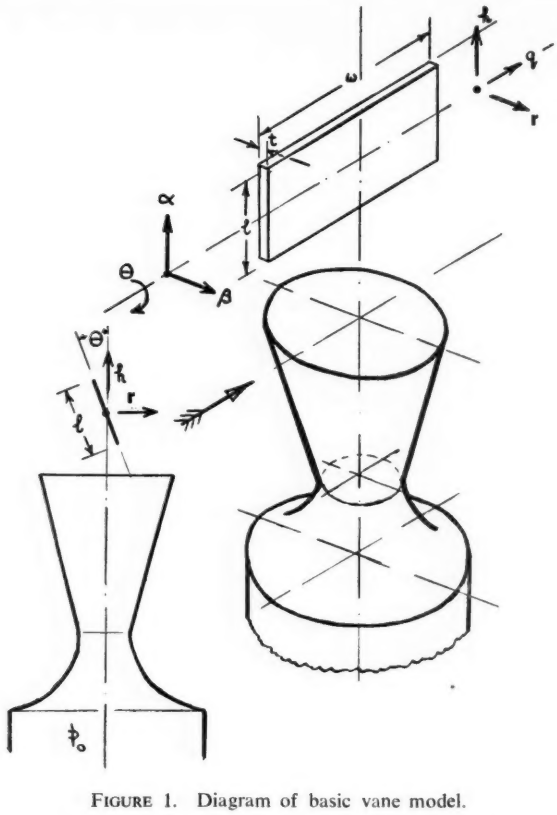


FIGURE 1. Diagram of basic vane model.

restricted to rocket-boosted aircraft. The motor alignment does not need to vary unless boost is required to aid high speed manoeuvre. Factors which make the design of a compact and light unit more difficult are that the combustion chamber must be mounted on four very robust gimbal mounted bearings which share between them the full load of the motor thrust. Liquid fuel motors require that the fuel supply lines to the combustion chamber be flexible. Thus the alignment of the whole combustion chamber is usually limited to large units.

Nozzles swivelling with respect to the combustion chamber by the use of ball-joint constructions present special difficulties under operating conditions, but permanently off-set nozzles are practical and have been employed in a number of experimental rocket designs and A.T.O. units.

2.2. AUXILIARY SIDE JETS

The control forces are readily predicted from design data, but details of construction may present some difficulty. The auxiliary side nozzles may be supplied with gases from either the main combustion chamber or from a separate generating unit. In the first case valves will be required to control the very high temperature gases from the main combustion chamber. With a separate supply unit the valve problem can be less severe because the gas generator can operate with a different fuel having lower temperature reaction. Auxiliary gas supply carries the penalty that some of the auxiliary fuel may never be used.

2.3. COMPRESSION DEFLECTION

If the jet carries, or can be made to carry, a net electrical charge, it might be deflected by applying a strong magnetic or electrical field. The electrical charge in rocket exhausts is relatively small and large magnetic or electrical forces would be necessary to produce appreciable deflection of the gas. It may be possible, however, to charge the gas deliberately and thus reduce the field forces required[10, 16, 17].

Any attempt to deflect the jet once it has been discharged from the nozzle must result in recompression with the attendant shock waves and rise in entropy. Thus the whole of the deflecting force must be taken up by the deflecting member and solid surfaces appear to be the only useful method of producing compression deflection.

All forms of deflection by solid surfaces or vanes can be related to the basic model illustrated in Fig. 1. The position of the vane may be varied with respect to the nozzle along the three axes h, r and q. It may be inclined at any angle θ about an axis which may be made to vary relative to the vane along the directions α and β. The vane has dimensions: span, w, chord, l and thickness, t. Finally, its shape in any plane may vary. Some vane arrangements derived from the basic model are shown in Fig. 2.

If the vane divides the jet equally it may be called a "centre vane" (Fig. 2(*a*)). Two crossed centre vanes can form an intersecting pair (Fig. 2(*b*)). If the centre vane is withdrawn along its longitudinal axis until only a portion of the end is within the stream, a "partially immersed vane" results (Fig. 2(*c*)). If the vane is moved at right angles to its longitudinal axis until it becomes a tangent to the jet boundary, it then produces the "paddle vane" (Fig. 2(*d*)). A simple modification of the paddle vane is to curve it so that it follows the jet boundary (Fig. 2(*e*)). This leads to the limiting case of the two opposite curved vanes meeting to form an enveloping tube which constitutes the "tubular extension" design (Fig. 2(*f*)).

2.4. EXPANSION DEFLECTION

Asymmetric expansion within the divergent part of a nozzle gives a deflected jet. Thus the control is exercised within the nozzle itself. No external force may be required and there may be no energy losses. Thus this method is attractive as a means of jet control although its realisation in practice is not simple.

It is simple to imagine a distorted nozzle which will permit unequal expansion about the centre line of the undistorted nozzle and so produce an asymmetrical jet with virtually perfect expansion. However, such a device does not allow change in direction during operation as the nozzle walls must be rigid.

Where a permanent deflection is required, nozzles designed to allow expansion deflection may find a useful application. It is possible to design such nozzles for perfect flow.[7] Fig. 3 shows a simple way of causing a non-adjustable asymmetric expansion by bevelling the exit plane of a conventional nozzle. This design is well known to turbine engineers[18]. So far little work appears to have been done to relate the jet angle to the

nozzle geometry[19, 20], although some measurements have been made with subsonic jets[21]. A modification of this method is illustrated in Fig. 3(*b*). The gas is under-expanded in a nozzle fitted with a cylindrical extension having an inclined exit plane called an ungular surface. Some expansion deflection occurs but also some compression deflection at the opposite side. Thus, the method will not be ideally isentropic but the losses may be small.

Expansion deflection can be brought about by the more subtle method of controlling the position of separation of the jet from the nozzle wall when it is not running full. Separation is very much influenced by the local conditions in the boundary layer adjacent to the wall, and also by the pressure of the atmosphere and its access to the region of detachment. By controlling the boundary layer, separation can be delayed (*c.f.* the prevention of breakaway from aerofoils). Applying control to the boundary layer on a limited circumferential arc will promote asymmetric separation and so cause expansion deflection. Boundary layer control can be effected by injecting gas through the nozzle wall or by removing gas by suction on such an arc. It could also be brought about by a modification of the flow of gas or liquid used for protecting the nozzle walls by transpiration or sweat cooling. It may also be practical to exercise some control over the access of air to the region of separation.

2.5. COMPARISON OF DIFFERENT METHODS OF JET CONTROL

Nozzle alignment offers a positive and predictable method of control that requires only small operating forces and, has the advantage that the nozzle thrust efficiency is unaffected. The method has been commonly adopted in practice, for example the Viking rocket[5] and in R.A.T.O. units[8].

Auxiliary side jets also produce a positive and predictable side thrust and require negligible operating force. If the energy is supplied from an auxiliary fuel it may carry the penalty of added weight, but it is the only method that can continue to operate after the main motor has been shut off or burnt out. Rocket "Redstone" employs side jets[22].

Compression deflection methods are positive and can be applied to existing vehicles without serious structural alteration. Centre vanes are subject to severe erosion and cause a drag loss whether in use or not. These disadvantages are avoided by the use of paddle vanes. The complete paddle vane or tubular extension is capable of producing high control forces but requires more operating power. Although vanes of one kind or another have been used in several practical designs[1, 2, 3, 4, 5, 6, 9, 12] no systematic information has been published and the data available refers in most cases to *ad hoc* tests. The experiments described later have been aimed to supply some correlated data for purposes of design.

Expansion deflection methods are elegant for it may be possible to use very small operating forces. In theory there need be no drag loss when the jet is

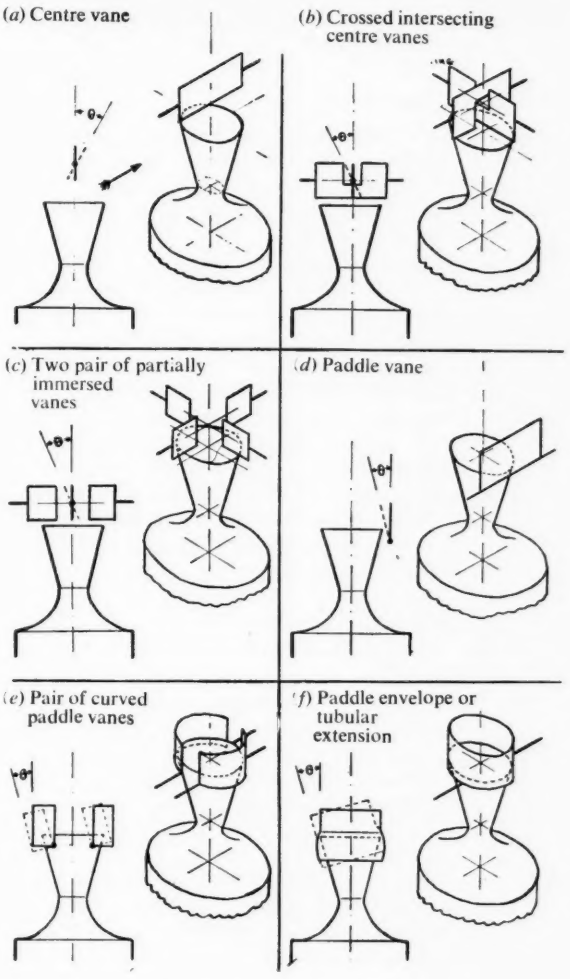
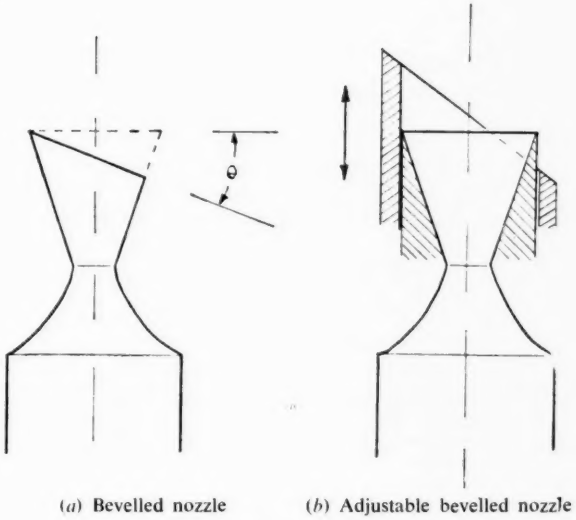

(a) Centre vane
(b) Crossed intersecting centre vanes
(c) Two pair of partially immersed vanes
(d) Paddle vane
(e) Pair of curved paddle vanes
(f) Paddle envelope or tubular extension

FIGURE 2. Different forms of vanes.



(*a*) Bevelled nozzle (*b*) Adjustable bevelled nozzle

FIGURE 3. Nozzle with bevelled exit plane.

deflected. Simple methods of variable expansion deflection are the subject of our present researches.

3. Experimentation

Since 1950, experimental work has been carried out at Imperial College into methods of jet control by compression and expansion deflection. The investigation has been confined to the control of jets produced by a model rocket nozzle of $\frac{1}{2}$ in. diameter throat discharging air to atmosphere from a reservoir at between 400 and 1,000 lb./in.^{2} and at room temperature. Details of the nozzle are shown in Fig. 4. It is designed to produce a jet of a Mach number of 2·81.

Experiments were made to measure the forces associated with deflection of the jet—the axial thrust of the nozzle, any perpendicular or side thrust resulting from deflection and, where appropriate, the control force needed to sustain the deflection. The forces were measured hydraulically and recorded photographically, and shadow photographs were taken of the deflected jet as an aid to interpreting the mechanism.

3.1. APPARATUS

The assembled apparatus is illustrated in Fig. 5. The axial thrust balance has been described in detail elsewhere[13].

This axial balance is compact, robust, simple to operate and easy to incorporate in other apparatus. The pressure gauge is the only component that requires calibration. When measuring axial reactions, an accuracy of the order of one per cent is possible but friction limits any improvement on this. The friction increases with non-axial reactions and the accuracy is correspondingly reduced, but adequate lubrication minimises this effect.

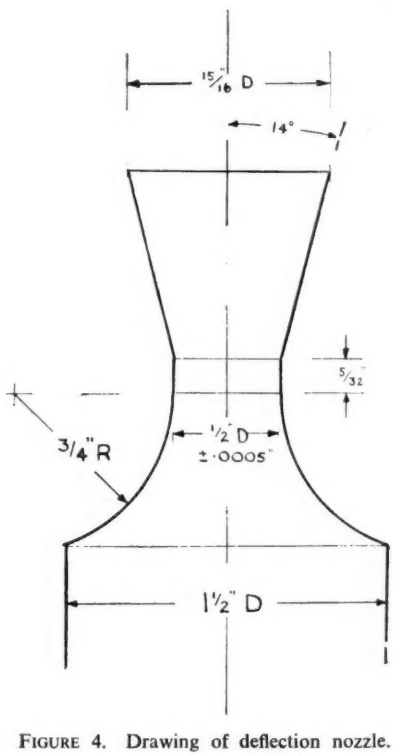


FIGURE 4. Drawing of deflection nozzle.

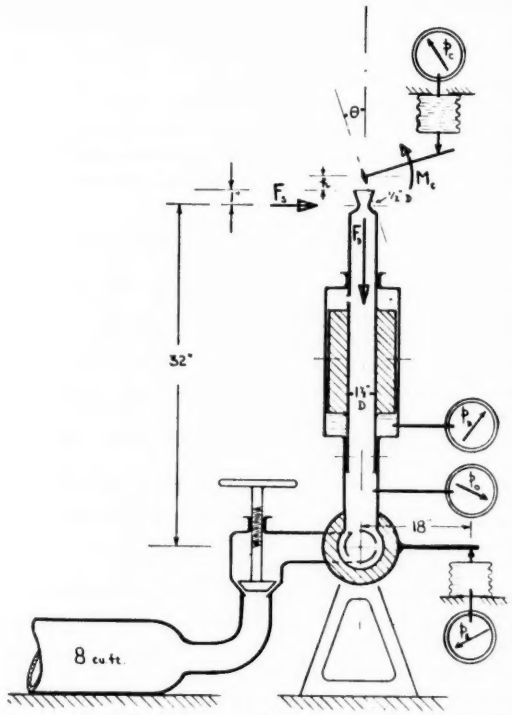


FIGURE 5. Diagrammatic sketch of assembled apparatus.

The axial thrust balance was mounted vertically in the free member of a frictionless rotary joint. The jet was deflected in a plane so that the whole assembly tended to rotate about the axis of the rotary joint but such movement was restrained by a lever arm bearing on a pressure recording element.

The pressure recording element was a metal capsule or bellows, oil filled and connected to a Bourdon gauge. The upper end of the bellows was rigidly attached to the lever arm and the lower end to the base plate.

In the general case of compression deflection, the deflecting member acting on the jet was hinged about an axis supported in ball races. A lever arm acted through a ball race onto another pressure capsule, with a screw adjustment to control the angle of the deflecting member.

The gauges recording axial thrust and reservoir pressure were calibrated regularly against a standard. The axial thrust was calculated from the dimensions of the balance and the recorded pressure. The side thrust and control force gauges were calibrated directly by suspending weights at a known leverage over the pressure capsules. Care was taken to detect hysteresis and every effort was made to minimise it. When hysteresis was observed, the calibration under conditions of falling load was used because this represented the conditions of the experiment.

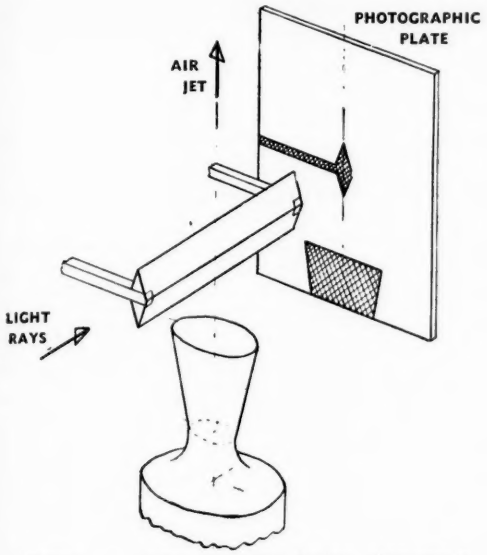


FIGURE 6. Arrangement for shadow photographs.

The photographic technique is fully described in Ref. 14. Fig. 6 shows the arrangement used to photograph the effect of a vane in a jet and Fig. 9 shows representative photographs.

3.2. EXPERIMENTAL DESIGNS

Each experiment was planned as a factorial design and the individual runs were always made in random order. All regression equations and the accuracies quoted are the result of systematic calculation from the whole body of experimental data. In some experiments, the technique of fractional replication was employed to economise in the number of runs. Details of the designs used are given in Ref. 15.

4. Compression Deflection—Vanes

4.1. EXPERIMENTAL PLAN

The number of variables associated with a vane control system is formidable, as Fig. 1 illustrates, and in a limited research programme some selection is necessary. Accordingly, this investigation was restricted to a study of the effects of the independent variables listed in Table I and illustrated in Fig. 7. The results of this experiment led to interest in the behaviour of vanes at small angles of inclination so that the work was followed by an investigation of the factors set out in Table II. In addition, tests were made to relate the two experiments completely.

TABLE I
EXPERIMENTAL CONFIGURATIONS—FIRST VANE EXPERIMENT

Height of Vane Axis above Nozzle Exit Plane, h	$\frac{1}{2}$ in. 1 in.
Vane Chord, l	$\frac{1}{2}$ in. $\frac{3}{4}$ in. 1 in.
Vane Angle, θ	5° 10° 15° 20°
Distance of Vane from Nozzle Axis, r	0 $\frac{3}{16}$ in. $\frac{1}{8}$ in. $\frac{3}{16}$ in.
Reservoir Pressure, p_0	400, 550, 700, 850, 1000 lb./in.²
Factorial combination of all above with $\frac{3}{4}$ in. chord repeated once — $\frac{1}{2}$ fraction of this total — 320 tests	

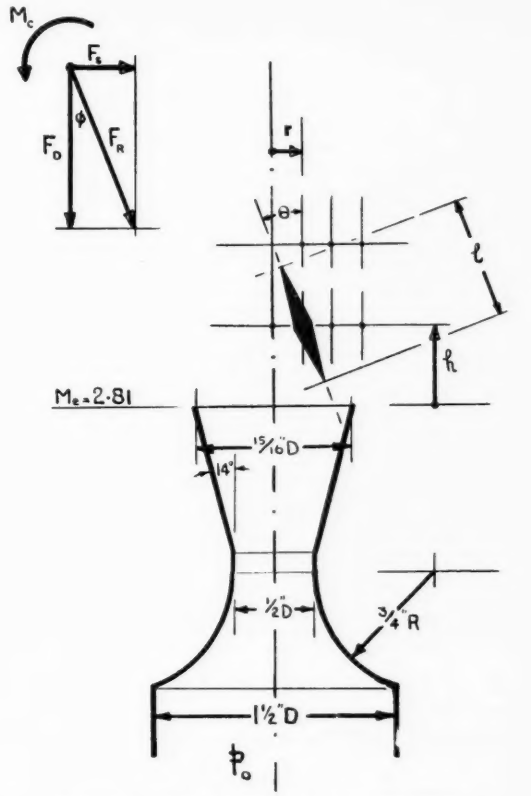


FIGURE 7. Experimental vane arrangements.

The vanes used were of double-wedge cross section and all of slenderness ratio, $t/l=\frac{1}{8}$. They were all 2 in. long, made of brass and finished with a high polish and were arranged so as to span the jet symmetrically.

The vanes were mounted in a carriage which clamped them at their ends and the whole carriage and vane could be rotated so that the axis of rotation passed through the centre of the cross section of the vanes. The bearing housings were fitted in trunnions which could be moved by screw mechanism in two directions (r and h in Fig. 1). The whole assembly was mounted in a suitable frame over the nozzle and the positions marked precisely.

The vane angle was measured on a protractor in the earlier work with an overall accuracy of about $\pm 1/3°$. In the later work at small angles, a much enlarged scale was used permitting an accuracy of about $\pm 1/10°$. Care was taken to locate the zero truly within this limit.

TABLE II
EXPERIMENTAL CONFIGURATIONS—SECOND VANE EXPERIMENT

Height of Vane Axis above Nozzle Exit Plane, h	$\frac{1}{2}$ in.
Vane Chord, l	$\frac{1}{2}$ in. $\frac{3}{4}$ in. 1 in.
Vane Angle, θ	0° 2° 4° 6° 10° 15°
Distance of Vane from Nozzle Axis, r	0 $\frac{3}{16}$ in. $\frac{1}{8}$ in. $\frac{3}{16}$ in.
Reservoir Pressure, p_0	700 lb./in.²
Factorial combination of all above repeated once— 144 tests	

4.2. EXPERIMENTAL RESULTS

4.2.1. *The Side Thrust of an Off-Centre Vane at Zero Incidence*

In an infinite supersonic field, a vane at zero incidence to the direction of motion can produce no lift or side thrust. In the very limited field of a jet, however, this condition does not hold. A vane of practical form which divides the jet unequally produces some side thrust at zero incidence.

In Fig. 8, the circle of radius R represents a cross-section through a jet and VV a vane at zero incidence and distance r from the centre. The leading edge of the vane will be the apex of two shock waves extending across the jet in the region bounded by mm' and nn'. These shock waves will divide and deflect the jet in opposite directions equally. In an infinite field, further expansion waves and shock waves downstream of the leading edge would restore the stream direction, but in a jet the disturbance in a finite region is permanent and the original stream direction is not fully recovered. Thus, the side thrust produced by an off-centre vane at zero incidence will be proportional to the difference in the jet divisions:—in Fig. 8, to the area A less the segment S which is the rectangle $mm'nn'$ or the product, $4ry$.

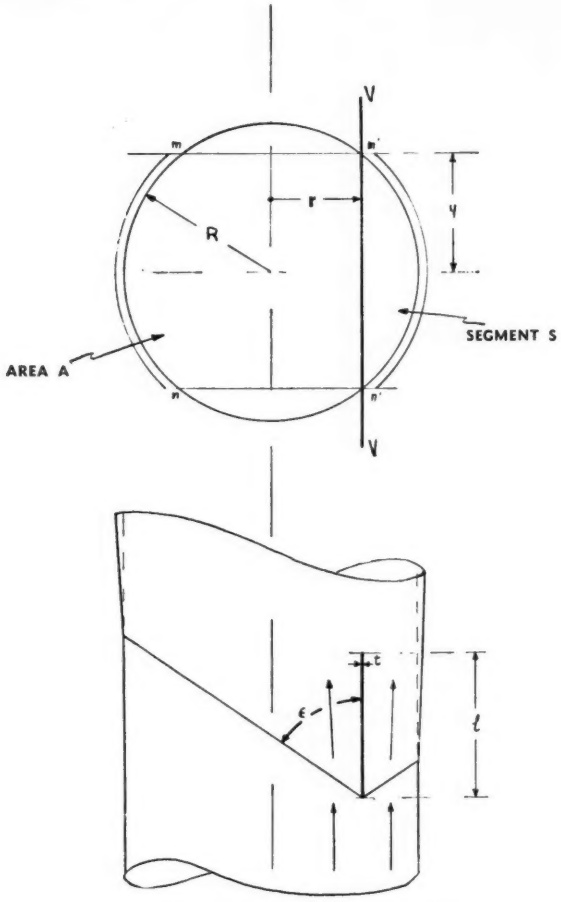


FIGURE 8. Diagram of an off-set vane.

A net deflection will occur when the jet passes the leading shock waves from an off-centre vane and an initially parallel jet will begin to diverge. This increase in jet diameter will reduce the possible recovery of direction when passing through the downstream shocks so that some net deflection will be maintained. The increase in jet diameter will vary with the vane chord and so the permanent deflection will increase with the vane chord, l, as well as the product $4ry$.

Shadow photographs of vanes of double wedge section in a supersonic jet confirm this general picture. Comparison of Fig. 9(*a*) and (*b*) shows that the jet spreads after passing the shock waves from the leading edge of the vane. The expected shocks appear at the trailing edge but do not restore the jet to its normal diameter. The angle of the leading shock, ε, is about 27°, which is close to the theoretical value of 26·3° for a stream at a Mach number of 2·81 and a vane of slenderness ratio, t/l, of 1/8. Fig. 10 shows the effect of traversing the jet with a vane. Deflection is plainly seen in (*b*) and the unequal recovery through the trailing edge shocks should be noted.

Measurements of side thrust with vanes at zero incidence show that this force varies with r as already described (Fig. 11) and is proportional to the product, $4ry$ and to the chord, l. Expressing the rectangular area $4ry$ as a fraction of the total cross-sectional area of the jet and the vane chord as a fraction of the jet diameter, the side thrust at zero incidence, $F_{s\,(0)}$ is given by

$$F_{s\,(0)} = K \frac{4ry}{\pi R^2} \frac{l}{2R} \quad . \quad . \quad . \quad (1)$$

The constant of proportionality, K, for $\frac{1}{2}$ in. nozzles has been determined experimentally and found to be 21·2 within the limits of our experimental error, ±0·44 lb. The effective jet radius, $R = 0{\cdot}539$ in. has been chosen to fit the data best when $p_0 = 700$ lb./in.^{2} (mean reservoir pressure) and is in accord with the rather crude measurements possible from shadow photographs.

The graph of Fig. 11 shows that the error in predicting the zero angle side thrust of the vane near the boundary, *i.e.* a paddle vane, may be very high. As the vane approaches the boundary, side thrust becomes very sensitive to small changes in vane position and to changes in the effective jet diameter.

4.2.2. *Deflection by an Inclined Vane*

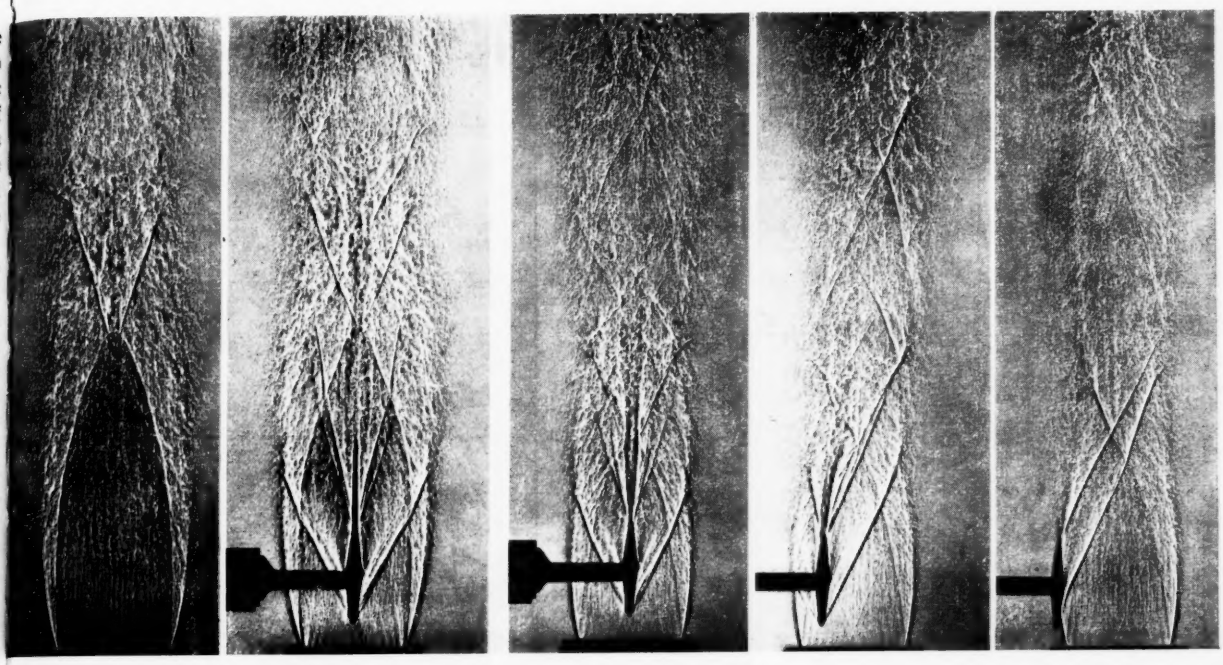
The side thrust or lift produced by an inclined vane of double wedge cross section in an infinite supersonic stream is given by,

$$F_s = (\tfrac{1}{2}\rho u^2)\, A.\, C_L \quad . \quad . \quad . \quad (2)$$

where the coefficient of lift is given by,

$$C_L = \frac{4\theta/57{\cdot}3}{\sqrt{M_\infty{}^2 - 1}} \quad . \quad . \quad . \quad (3)$$

For the purpose of this analysis, consider the jet to be an infinite stream. Then, for the nozzle used in this



(*a*) (*b*)

FIGURE 9. Perpendicular jet spread caused by a vane.

(*a*) (*b*) (*c*)

FIGURE 10. Flow pattern past off-centre vanes.

work (Fig. 4), the jet characteristics are such that

$$F_s = 0{\cdot}005333\, p_0 A\theta. \quad . \quad . \quad (4)$$

There is no problem in defining the area of a vane in an infinite stream. Where the vane is immersed in a jet and supported externally, the effective area will be defined as the product of the wetted span, $2y$, and the vane chord, l, (Fig. 8).

The side thrust of a centre vane has been experimentally determined and can be expressed by,

$$F_s = 0{\cdot}003161\, p_0\, A\, \theta. \quad . \quad . \quad (5)$$

This expression and the data are plotted in Fig. 12. Compared with equation (4), the factor in this empirical expression indicates an effectiveness of 60 per cent.

When the vane is off-centre and inclined in a jet, the side thrust resulting from its off-centre position must be added to the side thrust produced by vane inclination. Thus, the general expression for the side thrust set up

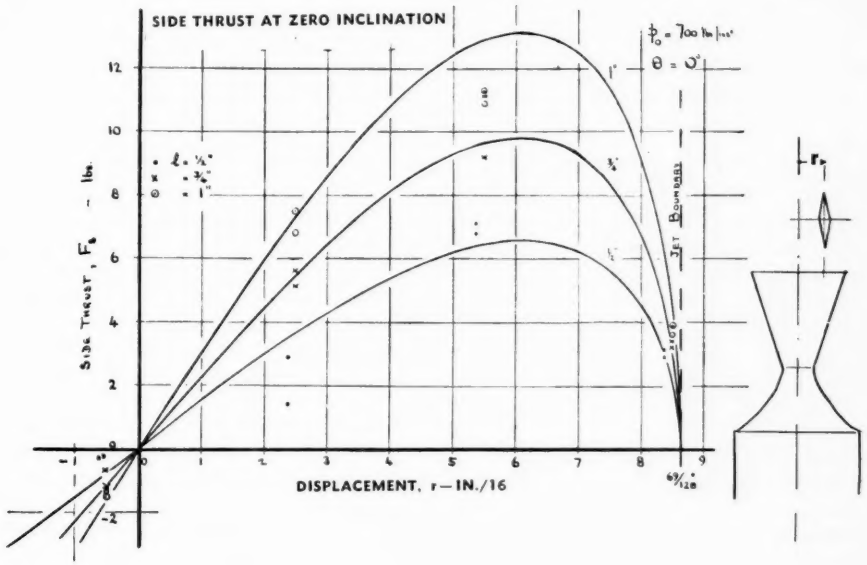


FIGURE 11. Side thrust at zero incidence.

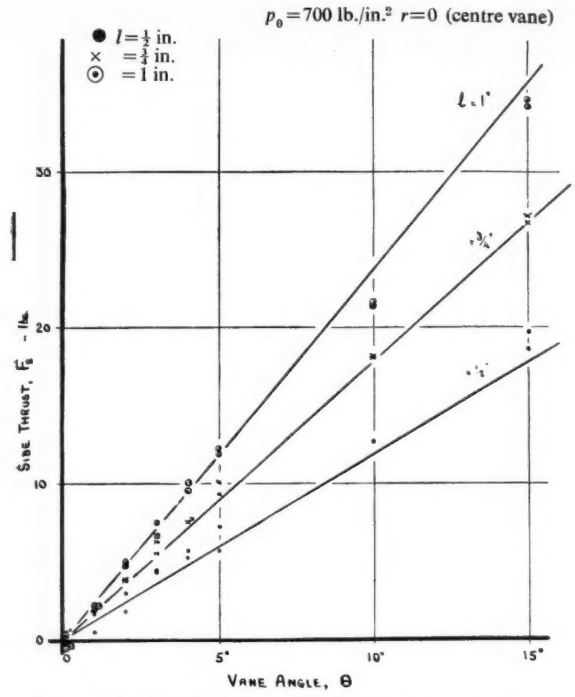


FIGURE 12. Variation of side thrust with vane angle—centre vane.

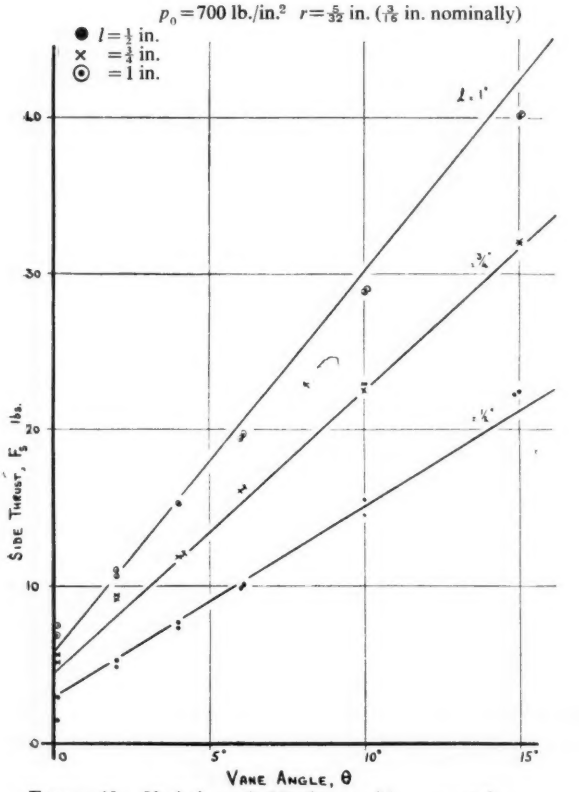


FIGURE 13. Variation of side thrust with vane angle—5/32 in. off-centre.

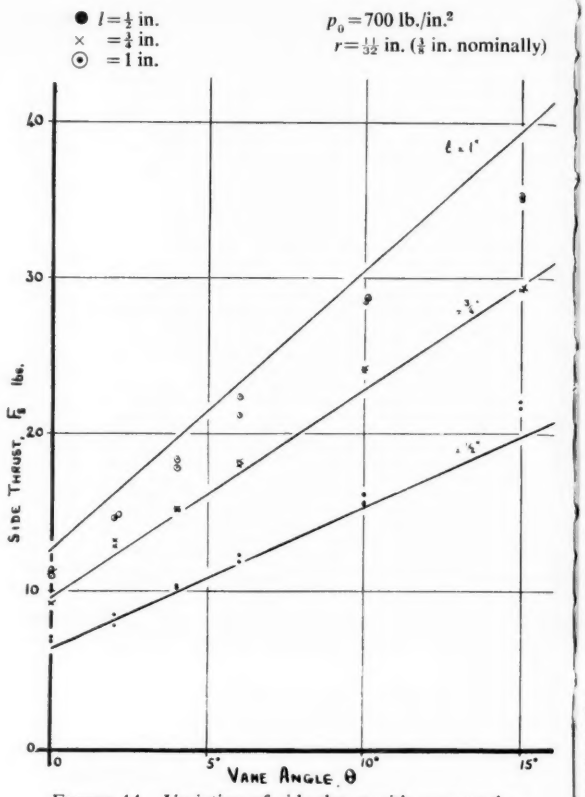


FIGURE 14. Variation of side thrust with vane angle—11/32 in. off-centre.

by a vane in any position is,

$$F_s = K\, p_0 A\theta + F_{s\,(0)}. \qquad . \quad . \quad (6)$$

The factor K has been found to be roughly constant (±3 per cent) at 0·003191 over the centre 2/3 traverse of the jet. This indicates a mean effectiveness of 60 per cent (Figs. 13 and 14).

The curves drawn in Figs. 12-14 are those represented by equation (6) where K takes its mean value, 0·003191. The experimental observations are also shown on the graphs and in one or two cases there appears to be a systematic departure from the curve. This deviation is not significant when compared with the experimental error (±0·44 lb.) and it must be kept in mind that the equation is based on the whole body of experimentation and not any particular group. The establishment of a simple unifying equation is a tremendous simplification in presenting the data and a valuable aid to predicting the behaviour of vanes in other jets.

In the case of a paddle vane, the jet boundary will be affected by the presence of the vane, the jet will spread over the inner face of the vane and the outer face will be unwetted so that the simplifying assumptions made cannot be expected to hold. The vane is no longer immersed in a supersonic field. The side thrust produced by a paddle vane, therefore, is difficult to predict. The side thrust at zero incidence will be uncertain because of the problem of defining the

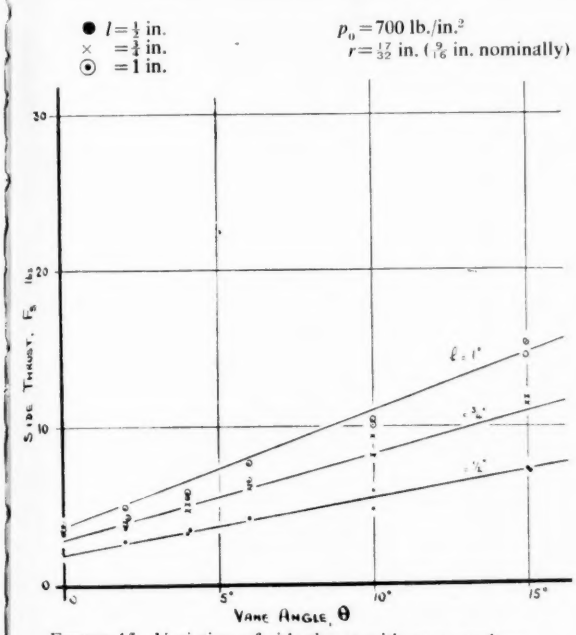


FIGURE 15. Variation of side thrust with vane angle—paddle vane.

geometry and the wetted area of the vane cannot be specified precisely. In this investigation a paddle vane at position $r=0{\cdot}531$ in. (98·5 per cent of the effective jet radius) showed an effectiveness of 101 per cent when compared with the theoretical equation (4). This is seen in Fig. 15.

Comparison with Figs. 13 and 14 shows the smaller side thrust produced by paddle vanes because of the reduced wetted area. It will be seen that the side thrust remains linear in vane angle as it does with other vanes. The order of magnitude of side thrust that may be expected has been indicated, but the difficulties of defining how the vane and the jet will interact leads to difficulty in predicting performance. Doubtless this problem explains the lack of agreement between the performance of the various *ad hoc* paddle vane designs that have been reported.

The effect of moving the vane away from the nozzle exit plane is very small—an average reduction of 5 per cent for the half inch movement examined.

4.2.3. *The Loss of Axial Thrust Caused by a Vane*

The loss of thrust caused by a vane, that is, the difference between the recorded axial thrust of a deflected jet and the thrust of an unobstructed one, is equivalent to the drag on the vane.

The drag on a double wedge vane in an infinite supersonic field is given by,

$$D=(\tfrac{1}{2}\rho u^2)\,A\,C_D. \quad . \quad . \quad . \quad (7)$$

where the drag coefficient is given by,

$$C_D=\frac{4}{\sqrt{M_\infty{}^2-1}}\left\{\left(\frac{\theta}{57{\cdot}3}\right)^2+\left(\frac{t}{l}\right)^2\right\}. \quad (8)$$

Again, treating the jet as an infinite field,

$$D=(0{\cdot}9312\theta^2+47{\cdot}77)\times 10^{-4}\times p_0 A. \quad . \quad (9)$$

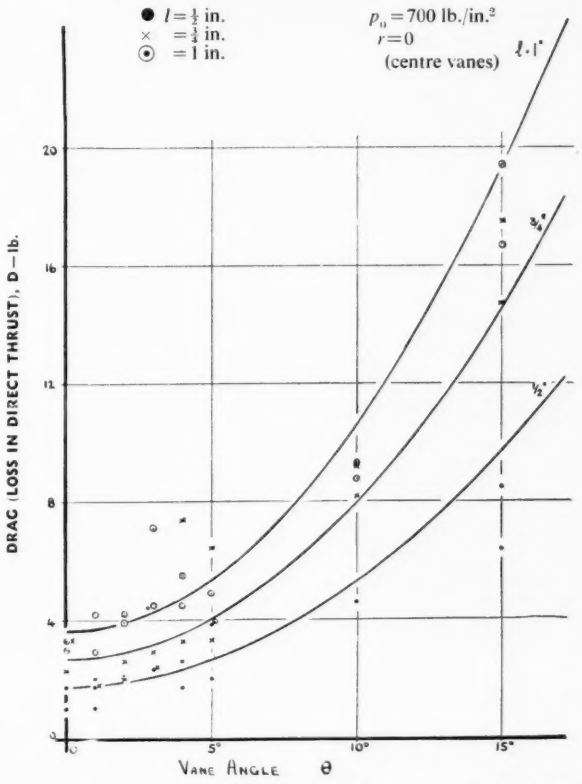


FIGURE 16. Variation of loss in axial thrust with vane angle, centre vane.

The experimental values show good agreement with the theoretical equation (9) for the case of a centre vane (Fig. 16). The apparatus was not ideal for measuring small differences in thrust when the load is not axial so that the experimental error is rather high ($\pm 1{\cdot}5$ lb.). The analysis of thrust loss has not been extended to off-centre vanes because it is felt that these, having a permanent side thrust at zero incidence, are of little practical interest. Qualitatively, it is found that the drag is roughly proportional to the side thrust so that it also passes through a maximum value as the vane traverses the jet.

4.2.4. *The Control Moment and the Centre of Pressure of the Vane*

In all cases, the control moment has been found to vary in a most complex manner with the independent variables. This is especially true in the case of off-centre vanes.

It is found that the control moment of a centre vane can be expressed simply by,

$$M_c=(0{\cdot}42\theta+5{\cdot}3)\times 10^{-4}\,p_0 \pm 0{\cdot}073 \text{ lb. ft.} \quad (10)$$

This is plotted with some of the data in Fig. 17.

This simple relationship has no theoretical background. The calculation of the centre of pressure from these data (*see* Appendix I for method) leads to a position independent of the vane chord and ahead of the leading edge for some configurations. However,

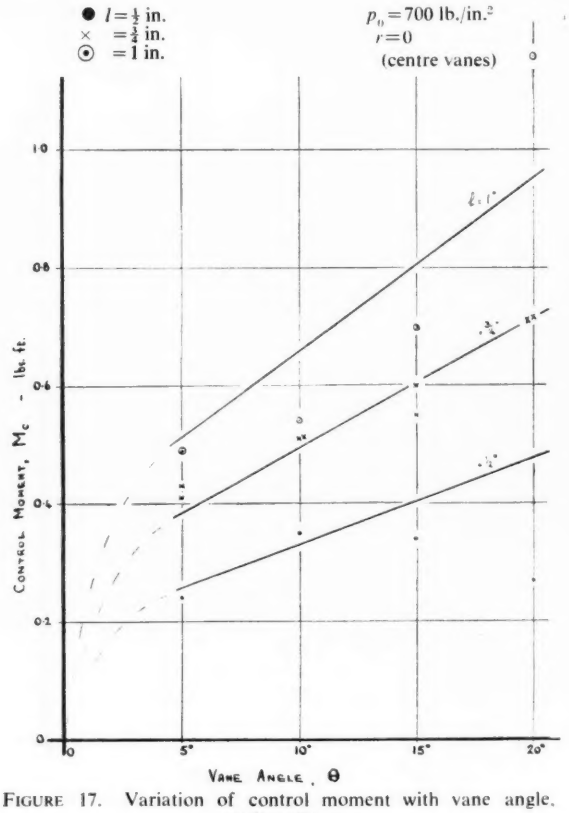


FIGURE 17. Variation of control moment with vane angle, centre vane.

this analysis does illustrate that the position of the centre of pressure varies considerably with vane angle, but the error is very high in this derived quantity ($\pm 0{\cdot}08$ in.) (Fig. 18).

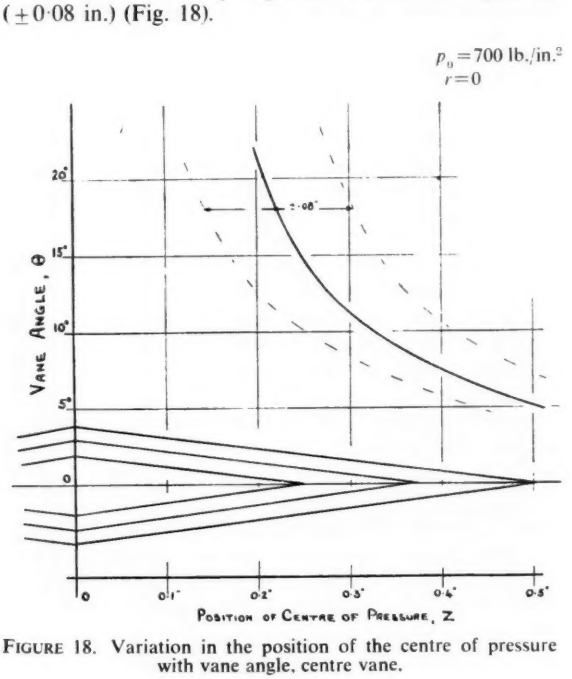


FIGURE 18. Variation in the position of the centre of pressure with vane angle, centre vane.

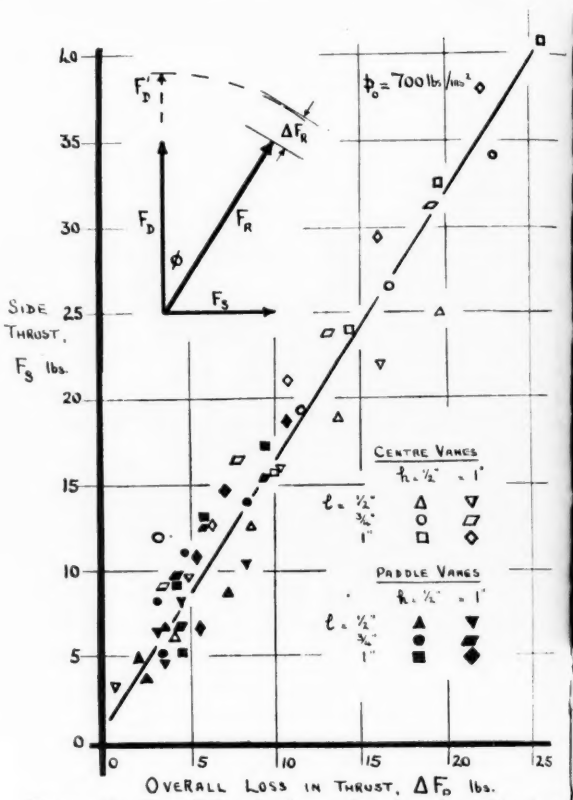


FIGURE 19. Overall loss in thrust plotted against side thrust.

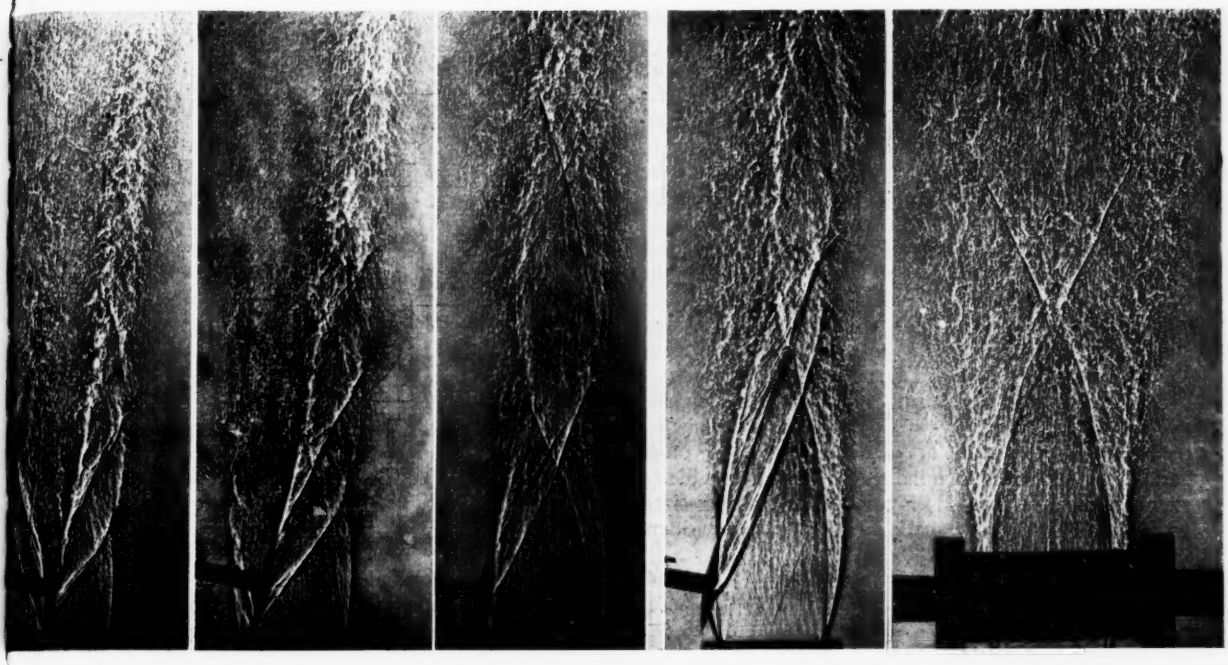
4.3. DISCUSSION OF THE RESULTS

The resultant thrust of a jet deflected by a vane, F_R, can be calculated from the measured side thrust, F_S, and the reduced axial thrust, F_D. When this resultant thrust is compared with the axial thrust of an unobstructed jet, F'_D, an overall loss in thrust is observed, ΔF_R (Fig. 19).

If for any vane arrangement the side thrust produced is compared with the resulting overall loss in thrust, ΔF_R, the ratio is found to be approximately constant for all vane configurations. This is shown by the straight line in Fig. 19, from the slope of which it can be seen that the overall loss averages 0·63 of the side thrust produced. This overall loss may be considered to be the cost of obtaining side thrust with thin vanes. The constant ratio shows that there is no optimum vane configuration which leads to "cheap" side thrust.

The pattern of flow around the vane is very much as expected from theory, except that there is a large unwetted area on the upper surface of the vane by reason of separation from the top surface (Fig. 20(*a*) and (*b*)). The angle of the leading shock in these photographs is in good agreement with that predicted by the two-dimensional Prandtl-Meyer theory.

A jet spreads across the face of the vane that deflects it and the spreading is greatest for a paddle vane as illustrated in Fig. 21. The vane spreads only a small circular segment of the jet. This is apparent from



(*a*) (*b*) (*c*)

FIGURE 20. Flow pattern past different vanes.

(*a*) (*b*)

FIGURE 21. Surface spread of jet over a paddle vane.

Fig. 21(*b*) where it is observed that the upper shock wave does not extend to the boundaries of the apparent jet. It has already been shown that the deflected jet conforms to two-dimensional theory. From the mechanical standpoint, the increased width of the stream about the vane will require that the vane and its supports be wider than the unobstructed jet. The problem of jet spreading may become more severe at high altitudes where under-expansion of the jet may occur[23].

Our results for vanes are in general agreement with the scant information available from other (mainly *ad hoc*) investigations. Disagreement occurs for paddle vanes and our experiments have shown that this arrangement is particularly sensitive to very small changes in the vane and jet configuration.

To summarise: the side thrust developed by a vane can be predicted from two-dimensional theory for an infinite field reduced by a factor of about 0·60. The drag of a centre vane is as predicted by theory but the control moment is higher than theory suggests. The behaviour of a paddle vane cannot be predicted with confidence because of the difficulty of defining the vane and jet configuration.

5. Compression Deflection—Inclined Tubular Extensions

5.1. EXPERIMENTAL PLAN

A simple modification of the paddle vane is to curve it to follow the jet boundary, *i.e.* a curved paddle vane —Fig. 2(*e*). An obvious development of this idea is to continue the envelopment until it becomes a tube on the end of the nozzle. This device provides a positive means of deflecting the jet and achieves deflection without spreading the jet.

Two simple designs were produced and are illustrated in Fig. 22. The tubular extension is supported upon bearings at right angles to the jet axis, the bearings being rotatable about the axis. To achieve a reasonable gas seal between the extension and the nozzle, they abut into each other in the form of a ball and socket with a clearance. In these experiments, the control moment was recorded by attaching a lever arm to the tubular extension which recorded the force on a pressure capsule.

A factorial experiment was planned to relate the measured forces to the angle of inclination and tube length for the two types. The experimental combinations are set out in Table III.

5.2. EXPERIMENTAL RESULTS

The side thrust obtained by this means of deflection in the case of design "*b*" (Fig. 22) can be expressed in a simple form by the empirical equation,

$$F_s = (0{\cdot}0038 + 0{\cdot}0036\, s - 0{\cdot}00115\, s^2)\, p_0 \theta \text{ lb.} \qquad (11)$$

TABLE III

EXPERIMENTAL CONFIGURATIONS—TUBULAR EXTENSIONS

Nozzle Design	Type "*a*"	Type "*b*"
Tube Length, s	$\frac{3}{4}$ in.	$1\frac{1}{4}$ in. $1\frac{3}{4}$ in.
Tube Angle, θ	5° 10° 15° 20°	
Reservoir Pressure, p_0	400, 500, 600, 700, 800, 900, 1000 lb./in.^{2}	
Factorial combination of all above with $1\frac{1}{4}$ in. tube length repeated once — 224 tests		

This formula expresses the experimental results within ± 4 lb. (*i.e.* about ± 5 per cent of the mean of the experimental results). Fig. 23 shows this approximate equation and the "most probable" results ($\pm 1{\cdot}1$ lb.) obtained from a systematic regression on all the data.

The side thrust is found to be not strictly linear in angle and the variation with reservoir pressure and tube length is quite complex in detail. The side thrust varies with the square of the tube length and there is an optimum length that gives the maximum side thrust for a given angle of deflection—about three times the nozzle throat diameter.

The behaviour of design type "*a*" (Fig. 22) is very similar to that of "*b*," but the average side thrust is slightly less at all corresponding values. This appears to indicate that the considerable difference in design at the nozzle exit plane has little influence in changing the degree or relation of the side thrust to other variables. With this in mind, it may be reasonable to suppose that equation (11) is applicable to larger sized nozzles. If this is so, equation (11) may be rewritten,

$$F_s = \left(0{\cdot}01936 + 0{\cdot}00917\left(\frac{s}{D_t}\right) - 0{\cdot}001477\left(\frac{s}{D_t}\right)^2\right) p_0 A_t \theta. \quad . \quad . \quad . \quad (12)$$

The axial thrust of a jet deflected by the tubular extension was found to be reduced by,

$$F_D = 10{\cdot}1 - 2{\cdot}26\theta + 0{\cdot}126\theta^2 \quad \pm \quad 2{\cdot}3 \text{ lb.} \quad . \quad (13)$$

at a reservoir pressure, $p_0 = 700$ lb./in.^{2} This amount is roughly equivalent to the drag of a centre vane of $\frac{1}{2}$ in. chord. No difference in loss was detected between designs "*a*" and "*b*." Some slight augmentation of thrust was observed at lower reservoir pressures and small angles in both cases. The tube length up to 4 throat diameters did not appear to affect the drag. This is probably because the major loss occurs in the entry section. Fig. 24 illustrates the way in which the extension tube recovers some of the divergence loss and this explains the thrust augmentation. Comparing

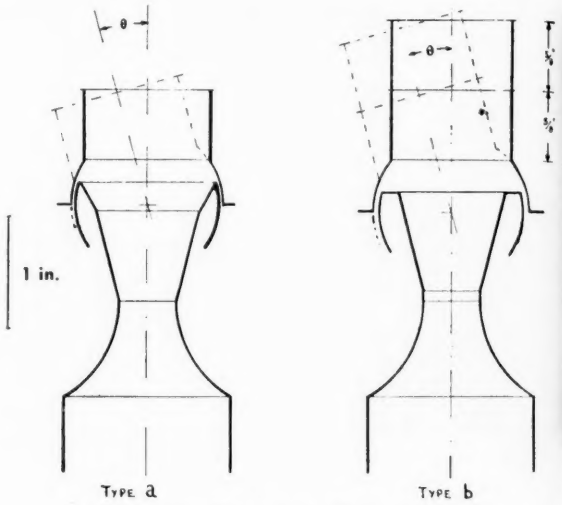


FIGURE 22. Drawing of tubular extensions.

photographs (*a*) and (*b*, *c*), the jet in (*a*) with no tube is divergent and in (*b*, *c*), with extension, convergent. When the tube is inclined at small angles (*f*), the jet is practically parallel and augmentation still occurs. At large angles (*e*) and (*g*), the jet has already separated but with the longer tube (*g*), it is still not seriously disturbed.

The control moment is found to vary in a complicated manner. There are numerous changes in the relationships between the independent variables shown in Table III. All relationships are found to be non-linear. There does not appear to be any optimum combination of the variables that gives a minimum control moment. The average control moment for type "*a*" was found to be higher than that for type "*b*."

As a guide to the order of magnitude,

$$M_c \rightleftharpoons 0{\cdot}00036\, p_0 \theta \text{ lb. ft.} \quad . \quad . \quad (14)$$

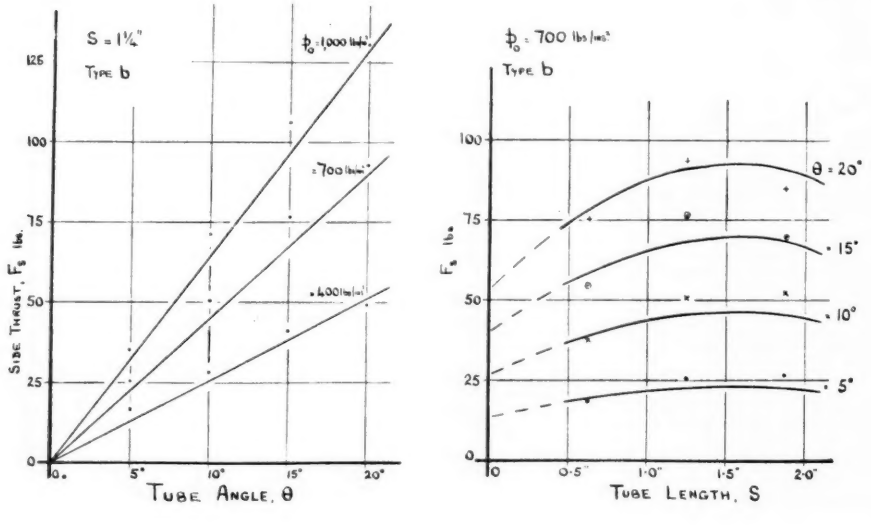
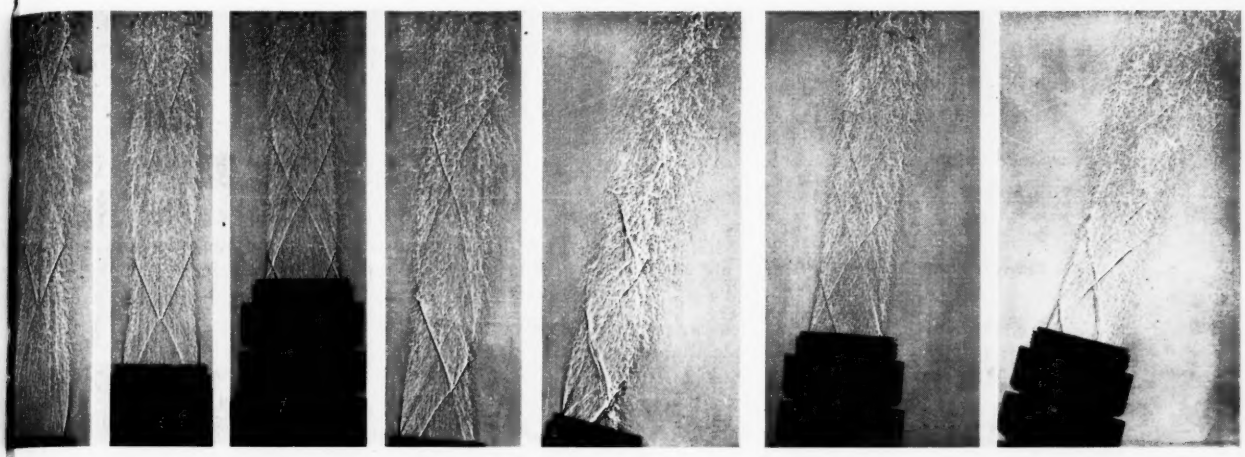


FIGURE 23. Variation of side thrust with angle and tube length.



(*a*) (*b*) (*c*) (*d*) (*e*) (*f*) (*g*)

FIGURE 24. Flow through tubular extensions.

From this approximate equation, the control moment necessary to move a tube is of the order of 0·05 lb. ft. per lb. of side thrust produced. The corresponding approximate figure for a centre vane would be 0·03 lb. ft. per lb. Judged, therefore, in terms of operating power, vanes are more economical than tubular extensions. On the other hand, for equal angle of inclination, the tubular extension gives a greater side thrust than a vane.

To summarise: inclined tubular extensions prove a positive and efficient means of jet deflection. The side thrust produced is roughly linear in angle and very large values can be obtained. The reduction in axial thrust is not very large but the control moment required is appreciably larger per unit effect than that of a centre vane. The optimum tube length is roughly $1\frac{1}{2}$ times the jet diameter.

6. Expansion Deflection

6.1. INTRODUCTION

An essential condition for effective expansion deflection is that the gases have not expanded correctly as they leave the walls of the nozzle. From this standpoint, the method is essentially inefficient but the loss of efficiency can be small. Of the methods of expansion deflection already discussed, only methods of bevelling the exit plane have been examined in this investigation. Here the gas must be above atmospheric pressure at the upstream end of the cut-away portion so that lateral expansion and hence deflection can occur.

6.2. NOZZLE WITH BEVELLED EXIT PLANE

An ordinary conical nozzle can be cut so that it has a bevelled exit plane. Fig. 3(*a*) shows details of the nozzle examined and the way it was bevelled progressively at 10° intervals from 0 to 50°. The side thrust and reduced axial thrust were measured at different reservoir pressures from 400 to 1,000 lb./in.^{2} The experimental configurations examined are listed in Table IV.

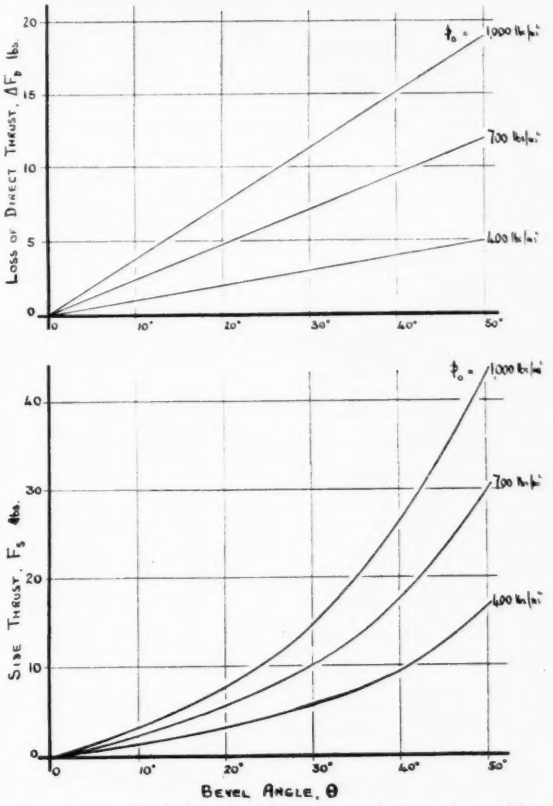


FIGURE 25. Variation of side thrust and thrust loss with bevel angle.

TABLE IV

EXPERIMENTAL CONFIGURATIONS—BEVELLED EXIT PLANE

Bevel Angle, θ	0° 10° 20° 30° 40° 50°
Reservoir Pressure, p_0	400, 500, 600, 700, 800, 900, 1000 lb./in.^{2}

Factorial combination of all above repeated once — 84 tests

The side thrust produced by a bevelled nozzle can be expressed by:

$$F_s = 0{\cdot}275\theta - 0{\cdot}0168\theta^2 + 0{\cdot}0000174p_0\theta^2 + \\ + 0{\cdot}000222\theta^3 \pm 0{\cdot}47 \text{ lb.} \quad . \quad . \quad (15)$$

The loss in axial thrust is given by,

$$\Delta F_D = (0{\cdot}000463p_0 - 0{\cdot}086)\theta \pm 2{\cdot}3 \text{ lb.} \quad . \quad (16)$$

These expressions are plotted in Fig. 25.

The angle the resultant thrust makes with the nozzle axis is given by,

$$\phi = 0{\cdot}00395\theta^2 - 0{\cdot}0125\theta \quad . \quad . \quad (17)$$

and the loss in total resultant thrust by,

$$\Delta F_R = -0{\cdot}113\theta + 0{\cdot}000428p_0\theta \text{ lb.} \quad . \quad (18)$$

A nozzle bevelled at 30°, for example, deflects its jet through about 4° and produces a side thrust similar to that of a typical vane at about 6°. These two examples will have very similar loss of axial thrust—about 2½ per cent of the normal thrust.

Figure 26 shows that the jet from a nozzle with a bevelled exit plane is relatively shock free and smooth, indicating that this method of control is simply expansion deflection. The angle of emergence at the point nearest the throat agrees closely with the predicted

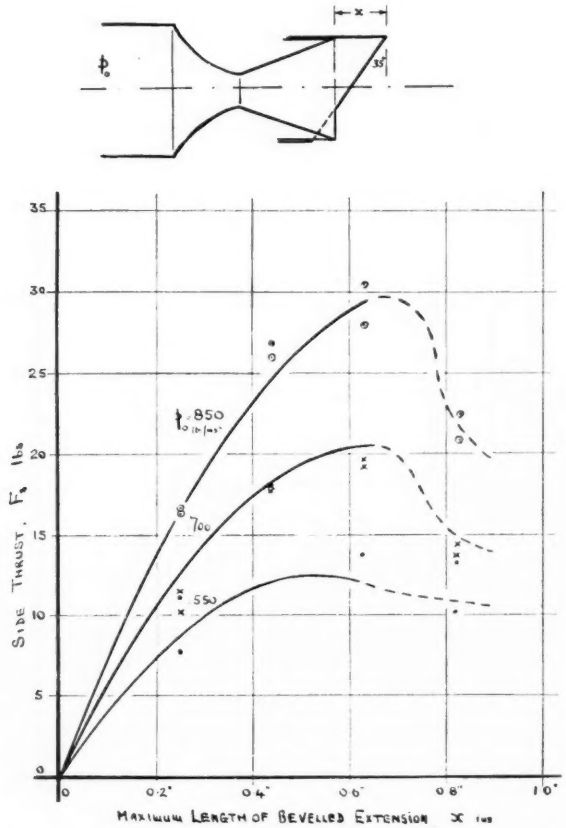


FIGURE 27. Variation of side thrust with length of bevelled extension.

TABLE V

EXPERIMENTAL CONFIGURATIONS—BEVELLED CYLINDRICAL EXTENSION (35°)

Maximum Extension	¼ in. 7/16 in. ⅝ in. 13/16 in.
Reservoir Pressure, p_0	550, 700, 850 lb./in.²
Factorial combination of all above repeated once — 24 tests	

Prandtl-Meyer angle for two-dimensional flow. The great difference between the emergence angle at the points farthest from, and nearest, the throats is clearly seen in the photographs.

6.3. NOZZLE WITH BEVELLED CYLINDRICAL EXTENSION

The bevelled cylindrical extension or ungular surface provides an effective method of producing side thrust[24]. The nozzle without the extension was designed to operate at a reservoir pressure of 308 lb./in.² When fitted with the extension, it was examined at 550, 700 and 850 lb./in.² The experimental configurations are listed in Table V.

The results are shown in Fig. 27. As the length of the extension is increased, the side thrust at first increases almost linearly and eventually reaches a maximum when the extension just fully envelopes the jet. Any further increase of the extension reduces the side thrust to a limiting value.

Up to the maximum value, the side thrust can be expressed by the equation,

$$F_s = -3{\cdot}4 + 0{\cdot}0033p_0 + 6{\cdot}6x + 0{\cdot}874p_0x - 53x^2 \pm 1{\cdot}0 \text{ lb.} \\ . \quad . \quad . \quad (19)$$

where x is the maximum length of the extension.

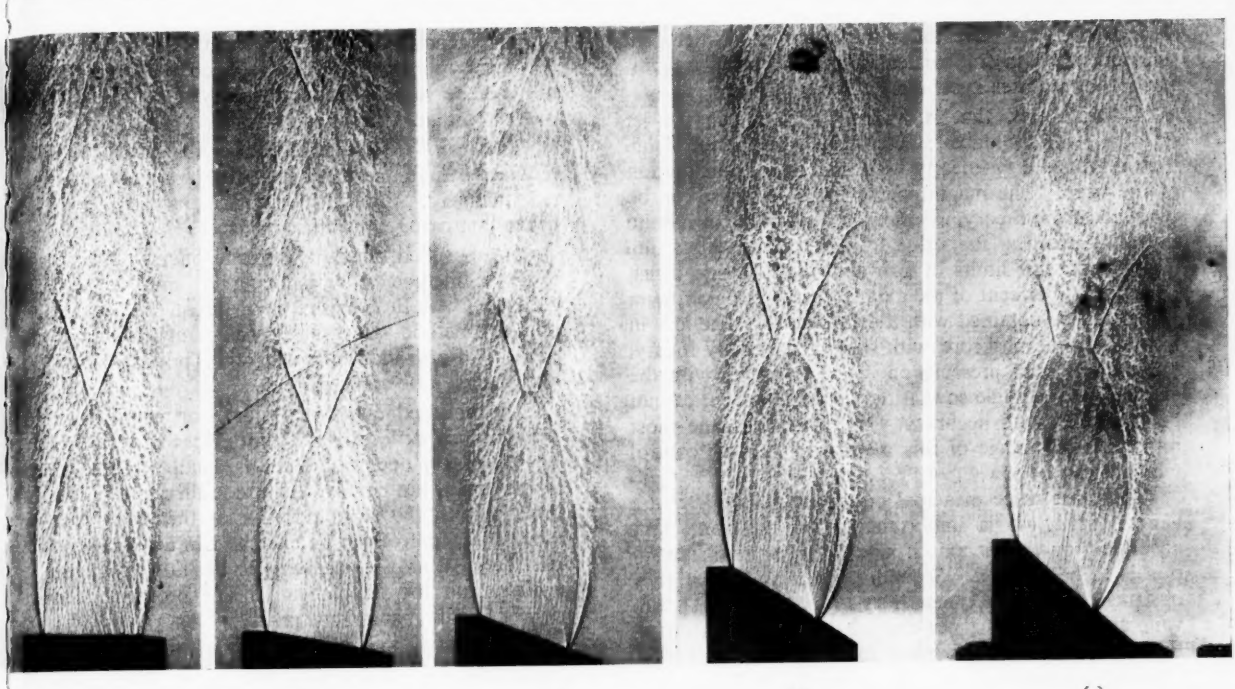
At a reservoir pressure about twice that for which the nozzle alone was designed, the maximum side thrust produced by a bevelled cylindrical extension is equivalent to that produced by a typical centre vane at 10° or by a typical paddle vane at 35°. At this reservoir pressure (*i.e.* operating the nozzle so that its jet is underexpanded), the loss in axial thrust when no deflection is imposed is about one per cent. As the curves of Fig. 27 show, the side thrust increases rapidly with the degree of underexpansion. A reservoir pressure twice the designed pressure appears to be a reasonable compromise that allows appreciable side thrust without excessive loss of axial thrust.

The loss in axial thrust accompanying deflection by a bevelled extension is very small. Its maximum value was only about 2½ per cent of the total thrust. This compares with a loss of 5 per cent for a centre vane or a paddle vane doing the same duty.

The shadow photographs of Fig. 28 show some shock waves in the jet from a bevelled extension so that the mechanism of deflection is not ideal expansion. None the less, the losses are very small. The effectiveness of this method of deflection indicates that curved paddle vanes will be efficient.

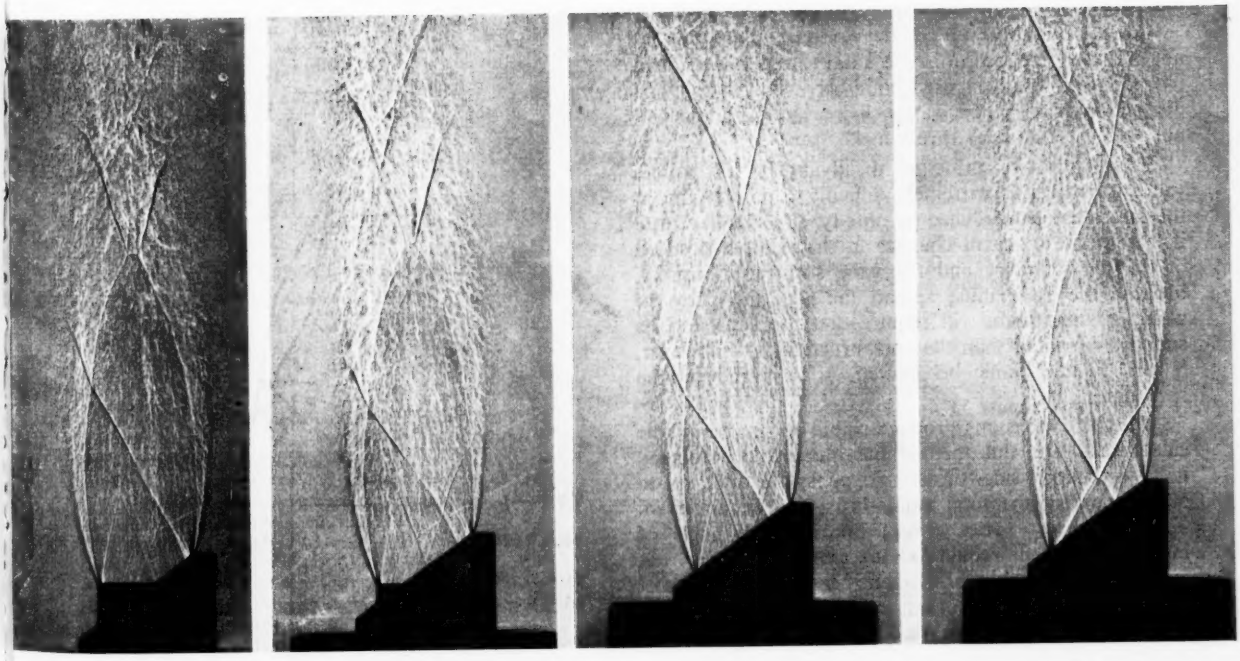
7. Design Considerations

This investigation has shown that there are a variety of practical ways of deflecting supersonic jets. The



(*a*) (*b*) (*c*) (*d*) (*e*)

FIGURE 26. Jets from nozzles of different bevel angles.



$x = \frac{1}{4}$ in. $x = \frac{7}{16}$ in. $x = \frac{1}{8}$ in. $x = \frac{1}{16}$ in.

$p_0 = 700$ lb./in.^{2}

FIGURE 28. Jets from a bevelled cylindrical sleeve.

various methods have different advantages which will recommend them for particular applications. A comparison of features is set out in Table VI.

Centre vanes, that is, vanes which divide the jet equally, will produce stable and reproducible side thrusts which vary linearly with the angle of inclination of the vane. The magnitude of the side thrust can be predicted from theory, assuming the jet to be an infinite field but reducing the value by a factor of about 0·6. Within practical limits of size and angle, a side thrust of about 20 per cent of the axial thrust is the maximum which can be obtained with a centre vane. The loss in axial thrust is in accord with that predicted by theory, but the centre of pressure on the vane moves with the change of vane angle so that the control moment cannot be estimated with accuracy. None the less, the most important advantage of this method is that the control moment can be low.

There are three principal disadvantages with centre vanes: firstly, it is important to set the vane very accurately on the jet axis if side thrust at zero inclination is to be avoided. Secondly, when not in use they obstruct the jet. A typical loss in axial thrust at zero vane angle is about 2 per cent of the total axial thrust. Finally, the vane is subjected to severe erosion.

Paddle vanes, that is, vanes breaking the boundary of the jet, also produce stable and reproducible side thrusts varying linearly with the angle of inclination. However, a paddle vane will produce only approximately one-third the side thrust of a similar centre vane. The reduction in axial thrust is directly proportional to the side thrust produced. In general, the control moment will be high because the position of the hinge axis cannot always be near the centre of pressure.

Some advantages of paddle vanes are that they do not cause any reduction in the axial thrust when not in use, nor are they exposed to erosive conditions. In distinction from centre vanes, increased thickness will not increase the axial thrust loss.

There are two principal disadvantages of paddle vanes: firstly, it is extremely difficult to predict any of the forces on, or produced by, this type of vane because it is sensitive to small changes in the relative position of the jet boundary and the vane surface. Secondly, because the vane must spread the jet, the increased width of the stream will require that the vane and its supports be wider than the unobstructed jet. However, if the paddle vane be curved, it can reduce the spreading.

Tubular extensions produce stable and reproducible side thrusts and this method has much to recommend it. Very large side thrusts are possible and up to 50 per cent of the total axial thrust can be achieved. The side thrust varies almost linearly with angle of inclination. The reduction in axial thrust is not excessive but the control moment per unit of side thrust produced is about twice that required for a centre vane hinged 50 per cent chord aft.

There are two notable advantages of tubular extensions: firstly there is no loss of axial thrust when the tube is not inclined Because the extension corrects the divergence loss, there may even be a gain in thrust. Secondly, the erosion of an inclined tubular extension may not be very severe as it is only in contact with the outer boundary of the jet.

The major disadvantage is that the control moment required is high and that this varies in a most complicated manner. Also, the tubular extension may be heavy compared with other devices.

The bevelled nozzle exit plane offers a simple and effective method of producing a small permanent deflection with little loss of axial thrust. The advantages of this method are its extreme simplicity and the absence of any additional weight. The disadvantage is that it cannot be adjusted in use.

The bevelled cylindrical sleeve or ungular surface will produce deflection with little loss of axial thrust. The necessary operating force is small and the weight of the device can be low. The side thrust varies almost linearly with the axial position of the sleeve and by rotating it, control can be exercised in any plane. The effectiveness of the sleeve will be little changed by erosion and the attachment is not prone to damage during handling and stowage.

The disadvantages of this method are that a permanent loss of one per cent axial thrust must be tolerated and, without increasing this loss seriously, side thrusts of more than 10 per cent of the axial thrust cannot be obtained.

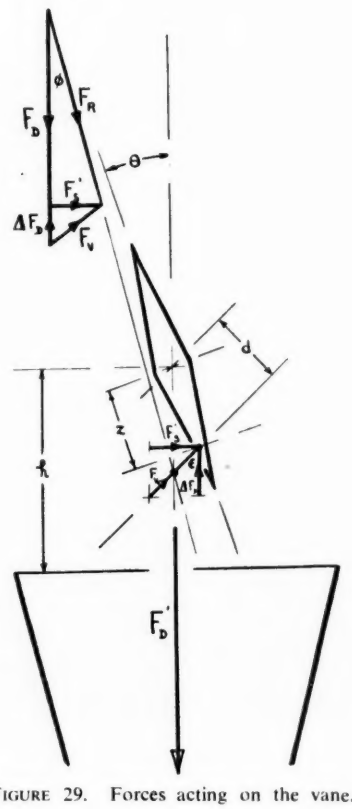


FIGURE 29. Forces acting on the vane.

TABLE VI

THE FEATURES OF DIFFERENT DEFLECTION METHODS

Feature	*Centre Vane*	*Paddle Vane*	*Tubular Extension*	*Bevelled Nozzle*	*Bevelled Extension*
Practical Max. Side Thrust	20%	10%	50%	5%	10% of the axial thrust
Loss of Axial Thrust when not in use	Mod.	Nil	Nil (Possibly Gain)	Nil	Small
Control Moment Required	Small	High	V. High	—	—
Erosion	High	Small	Small	Small	Small
Weight	Low	Low	Mod.	Nil	Low
Predictability	V. Good	V. Bad	Mod.	Good	Good

APPENDIX

Determination of the Centre of Pressure

Consider the total forces acting on the vane. In passing through the zone of influence exerted by the vane (within the bounds of its shock waves), some axial deceleration of the jet occurs and some acceleration takes place in a perpendicular direction. Expressed differently, there is a loss of axial thrust (drag force) and a gain of side thrust (lift). It is these forces that act upon the vane (Fig. 29).

If F_v is the resulting force acting on the vane and ΔF_D is the loss in axial thrust (= drag),

$$F_v^2 = F_s^2 + \Delta F_D^2 \quad . \quad . \quad . \quad (20)$$

Further,

$$\varepsilon = \tan^{-1} F_s / \Delta F_D \quad . \quad . \quad . \quad (21)$$

and the control moment will be given by,

$$M_c = F_v \times d \quad . \quad . \quad . \quad (22)$$

by geometry, d is related to z by,

$$z = \frac{d}{\cos\theta \cos\varepsilon (\tan\theta + \tan\varepsilon)} \quad . \quad . \quad (23)$$

Thus, the position of the centre of pressure can be determined from the measured side thrust and loss in direct thrust.

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TECHNICAL NOTES

Heat Transfer in a Laminar Boundary Layer with Constant Fluid Properties and Constant Wall Temperature

by

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A SIMPLE METHOD is given for the calculation of the heat transfer from a laminar flow surface. Computation is by a quadrature. The method is essentially a simplification and extension of the Eckert[1] method, and is applicable both to two-dimensional and to axisymmetric flows.

1. Introduction

Exact calculation of heat transfer between an aerofoil and a surrounding steady laminar stream involves the successive solution of two parabolic partial differential equations. The first, for the velocity field, is non-linear; the second, for the temperature field, is linear. The solution procedure is laborious and has rarely been attempted. Instead it is usual, for design purposes, to use approximate "profile" methods: assumptions are made about the "shapes" of the velocity and temperature profiles in the boundary layers, which reduce the partial differential equations to first-order ordinary differential equations. Two classes of procedure may be distinguished.

Class I procedures. Only one equation is solved. Eckert[1] solves the temperature equation by assuming that the rate of growth of the temperature layer is the same as for a "wedge" flow with the same temperature thickness and the same value of a certain pressure gradient parameter. Seban[2] and Drake[3] solve the velocity equation by a similar procedure and assume that the momentum thickness and the heat transfer rate are related in the same way as on wedges.

Class II procedures. Both equations are solved. The methods of Squire[4], Schuh[5], Lighthill[6] and Tribus and Klein[7] are of this type.

Class I procedures are simpler but less accurate; moreover they are restricted to the case that the aerofoil temperature is uniform from the leading edge backwards. When the wall temperature varies, or when the heated section does not start at the leading edge, Class II procedures must be used.

This note describes a simplification of the Class I method of Eckert. His graphical integration is replaced by a "quadrature" procedure similar to that used for momentum thickness calculations by Walz[8], Young and Winterbottom[9], Thwaites[10] and others. Two alternative quadratures are given, and the possibility of improvement of accuracy by an easily calculated "error" term is shown.

Notation

β parameter of "wedge" solutions
$\pi\beta$ semi-angle of wedge
c characteristic length of body: major axis for ellipse
Δ "temperature thickness"
Δ_1 $\int_0^\infty (1-\tau)\,dy$ = "temperature displacement" thickness
Δ_2 $\int_0^\infty \frac{u_1}{U}(1-\tau)\,dy$ = "enthalpy flux" thickness
Δ_4 $1\Big/\left(\frac{\partial\tau}{\partial y}\right)_0$ = "heat flux" thickness
Δ_m "mixed thickness"
Δ_n undefined thickness
h heat transfer coefficient
η variable in "wedge" solutions $= y\sqrt{\left(\frac{u_1}{\nu x}\frac{1}{(2-\beta)}\right)}$
k thermal conductivity of gas
ν kinematic viscosity
r radius co-ordinate of axisymmetric body
σ Prandtl number
τ dimensionless temperature on scale main stream = 1, wall = 0
u_1 main stream velocity, *i.e.* at outer edge of boundary layer
U approach velocity
x distance along surface from stagnation point
y distance normal to surface
Z dimensionless temperature thickness in wedge solutions $Z = \Delta\sqrt{\left(\frac{u_1}{\nu x}\frac{1}{(2-\beta)}\right)}$.

2. The "continuation equation" for the temperature boundary layer

Dimensional analysis shows that a differential equation for the growth of any temperature boundary thickness Δ may be written in the form

$$\frac{u_1}{\nu}\frac{d\Delta^2}{dx} = f\left(\frac{\Delta^2}{\nu}\frac{du_1}{dx}\right). \quad . \quad . \quad (1)$$

provided (*i*) the Prandtl number is constant, (*ii*) the rate of growth depends only on local conditions, and (*iii*) any dependence of the rate of growth on velocity layer shape or thickness is ignored. Equation (1) is an equation of Class I. It is obtained by treating the x, y, and z directions as independent dimensions, and incorporating the boundary layer assumptions that viscosity and conductivity respond only to gradients in the y-direction.

The unknown function in (1) we take from the exact "wedge" solutions (*see* Appendix 1). Figure 1 shows the appropriate functions for Δ_1, Δ_2 and Δ_4. As far as

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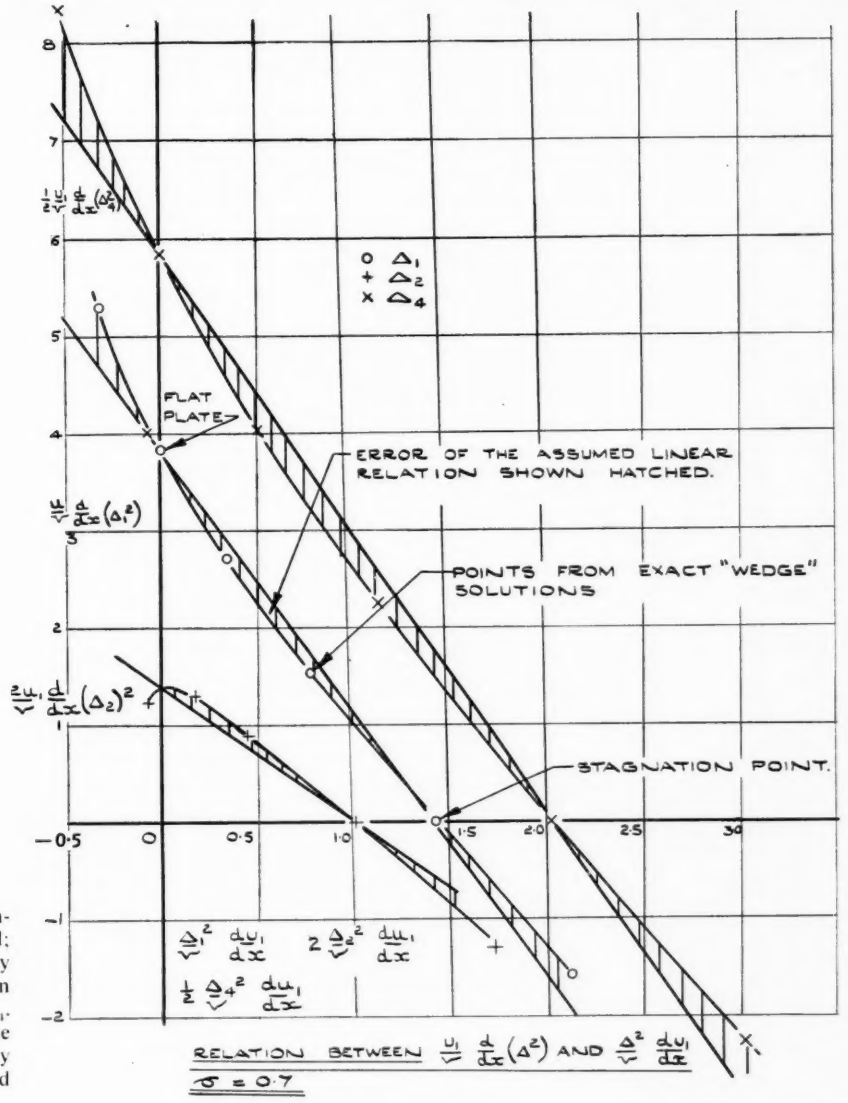

FIGURE 1.

equation (1) is concerned, any temperature thickness may be used; our choice will be governed by simplicity of integration. Attention will first be concentrated on Δ_1. Any process yielding Δ_4 will be convenient on account of the very simple relation between Δ_4 and the heat transfer coefficient,

$$h = k/\Delta_4 \quad . \quad . \quad . \quad (2)$$

Noting from Fig. 1 that the graph of $\frac{u_1}{\nu}\frac{d\Delta_1^2}{dx}$ against $\frac{\Delta_4^2}{\nu}\frac{du_1}{dx}$ is nearly straight, (1) is rewritten

$$\frac{u_1}{\nu}\frac{d}{dx}\left(\Delta_4\right)^2 = 11{\cdot}68 - 2{\cdot}87\left(\frac{\Delta_4^2}{\nu}\frac{du_1}{dx}\right) + E_4\left(\frac{\Delta_4^2}{\nu}\frac{du_1}{dx}\right) \quad . \quad . \quad . \quad (3)$$

where $E_4\left(\frac{\Delta_4^2}{\nu}\frac{du_1}{dx}\right)$ is the small vertical distance between the straight line and the curve in Fig. 1. Values of E_4 are:—

$\frac{\Delta_4^2}{\nu}\frac{du_1}{dx}$	6·05	4·07	2·26	1·01	0	−1·02
E_4	+1·042	0	−0·68	−0·67	0	+2·06
		↑ stagnation point			↑ flat plate	

Since E_4 is small it may be ignored when high accuracy is not demanded.

3. Quadrature formula for the temperature boundary layer thickness

The integrating factor $u_1^{1\cdot87}$ makes the sum of the left-hand and middle right-hand terms of equation (3) an exact differential. The equation may therefore be integrated as a quadrature, namely,

$$\left[\frac{u_1^{2\cdot87}}{\nu}\Delta_4^2\right]_0^x = 11{\cdot}68\int_0^x u_1^{1\cdot87}\,dx + \int_0^x u_1^{1\cdot87}\,E_4\left(\frac{\Delta_4^2}{\nu}\frac{du_1}{dx}\right)dx \quad (4)$$

From equation (4) Δ_4 may be calculated since u_1 is a known function of x for a given aerofoil and ν is known. The first approximation to Δ_4, which we call $\Delta_{4(1)}$, is obtained by ignoring the last term in (4), and the second approximation by inserting $\Delta_{4(1)}$ into this last term. Often the first approximation will suffice for design

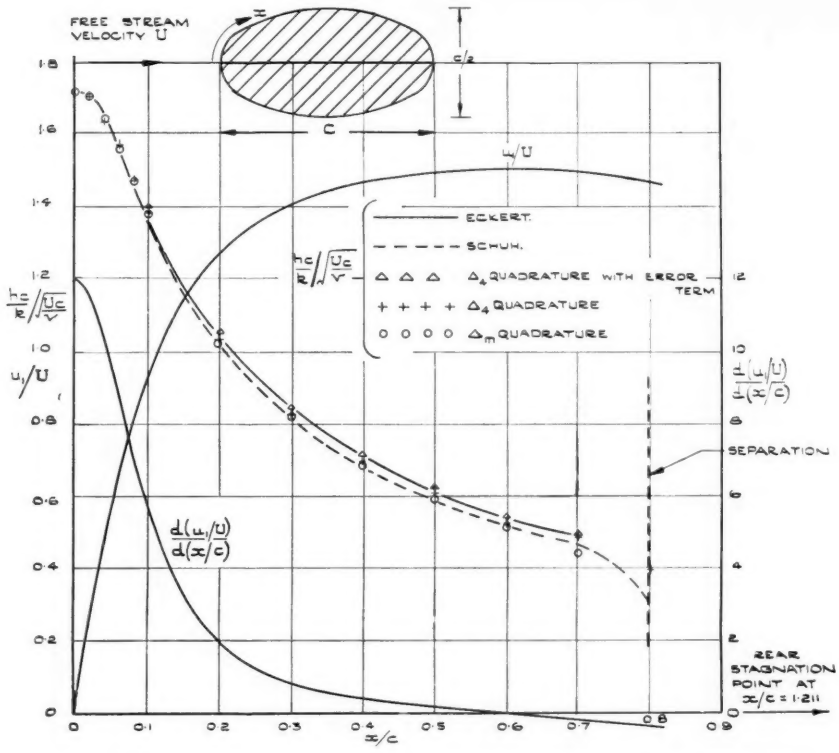

FIGURE 2. Results of different calculation methods for 2:1 ellipse.

purposes. With Δ_4 evaluated the coefficient of heat transfer is found from (2).

Usually dimensionless forms of (4) and (2) will be preferred. They are:

$$\left(\frac{\Delta_4}{c}\right)^2 \frac{Uc}{\nu} = \frac{11{\cdot}68}{\left(\frac{u_1}{U}\right)^{2{\cdot}87}} \int_0^{x/c} \left(\frac{u_1}{U}\right)^{1{\cdot}87} d\left(\frac{x}{c}\right) +$$

$$+ \frac{1}{\left(\frac{u_1}{U}\right)^{2{\cdot}87}} \int_0^{x/c} \left(\frac{u_1}{U}\right)^{1{\cdot}87} E_1 \left[\left(\frac{\Delta_4}{c}\right)^2 \cdot \frac{Uc}{\nu} \cdot \frac{d(u_1/U)}{d(x/c)}\right] d\left(\frac{x}{c}\right) \quad . \quad . \quad (5)$$

and $$\frac{hc}{k} \Big/ \sqrt{\left(\frac{Uc}{\nu}\right)} = \frac{1}{\left(\frac{\Delta_4}{c}\right)\sqrt{\left(\frac{Uc}{\nu}\right)}} \quad . \quad . \quad (6)$$

Equations (5) and (6) have been computed for Eckert's 2:1 ellipse and compared in Fig. 2 with his results and those of Schuh. The figure shows that the E_1 term contributes little to $(hc/k)/\sqrt{(Uc/\nu)}$. Numerical integration of (5) was made by Simpson's rule using 0·01 intervals to $x/c = 0{\cdot}1$ and then 0·05 intervals. Table I gives a summary of the computation. The use of Δ_1 instead of Δ_4 produces negligible change. At the front stagnation point, where

$$\frac{u_1}{U} = \left(\frac{x}{c}\right)\left(\frac{d(u_1/U)}{d(x/c)}\right)_0$$

equation (5) becomes

$$\left(\frac{\Delta_4}{c}\right)^2 \left(\frac{Uc}{\nu}\right) = \frac{4{\cdot}07}{\left[\frac{d(u_1/U)}{d(x/c)}\right]_0}$$

4. SIMPLER QUADRATURE USING A "MIXED" TEMPERATURE THICKNESS

Any temperature thickness may be used in equation (1). The essential feature which produces a quadrature is linearity of $f[(\Delta^2/\nu)\,du_1/dx]$. For shortness let $f[(\Delta_4{}^2/\nu)\,du_1/dx]$ be called f_4, and so on. Figure 1 shows the graphs of f_4 and f_2 to have opposite curvature, and this suggests that a "mixed" Δ_m, suitably combined from Δ_2 and Δ_4, might produce an f_m which is closer to linearity than either f_2 or f_4. A few trials showed that

$$\Delta_m{}^2 = \Delta_4{}^2 + 11{\cdot}82\,\Delta_2{}^2 \quad . \quad . \quad (7)$$

gave $f_m = 19{\cdot}78 - 2[(\Delta_m{}^2/\nu)\,du_1/dx]$ as the linear approximation. The resulting quadrature is

$$\left(\frac{\Delta_m}{c}\right)^2 \frac{Uc}{\nu} = \frac{19{\cdot}78}{\left(\frac{u_1}{U}\right)^2} \int_0^{x/c} \left(\frac{u_1}{U}\right) d\left(\frac{x}{c}\right) \quad . \quad (8)$$

In (8) the error term has been ignored since it was found to be only about one quarter of that for either the Δ_1 or the Δ_4 quadrature. Besides having a negligible error term, equation (8) is exceptionally easy to compute, since the integrand is the first power of (u_1/U).

The relation between Δ_m and h is not so simple as equation (2), being:

$$h = \frac{kC_m}{\Delta_m} \quad . \quad . \quad . \quad . \quad . \quad (9)$$

with dimensionless form

$$\frac{hc}{k} \Big/ \sqrt{\left(\frac{Uc}{\nu}\right)} = C_m \Big/ \left(\frac{\Delta_m}{c}\right)\sqrt{\left(\frac{Uc}{\nu}\right)}. \quad (10)$$

C_m depends on $[(\Delta_m{}^2/\nu)\,du_1/dx]$, and is given in Fig. 3. Its derivation is given in Appendix 2.

Results of a computation using equations (8) and (10) are given in Fig. 2. Numerical integration was made using the same intervals as for Δ_4, and a summary of the computation is given in Table II. At the front stagnation

TABLE I. Extract from calculations by Δ_1 method.

x/c	0	0·1	0·2	0·3	0·4	0·5	0·6	0·7	0·8
u_1/U	0	0·926	1·273	1·400	1·461	1·491	1·500	1·495	1·466
$I=\int_0^{x/c}(u_1/U)^{1\cdot87}\,d(x/c)$	0	0·0362	0·162	0·336	0·533	0·741	0·954	1·166	1·374
$\left(\frac{\Delta_1}{c}\right)^2\left(\frac{Uc}{\nu}\right)=11\cdot68\,I/(u_1/U)^{2\cdot87}$	0·339*	0·527	0·946	1·492	2·095	2·75	3·45	4·29	6·31
$\frac{hc}{k}\Big/\sqrt{\left(\frac{Uc}{\nu}\right)}$ (1st approx.)	1·716	1·378	1·030	0·820	0·690	0·603	0·538	0·483	0·398
$\left(\frac{\Delta_1}{c}\right)^2\left(\frac{Uc}{\nu}\right)\frac{d(u_1/U)}{d(x/c)}$	4·07	3·11	1·89	1·34	0·84	0·44	0	−0·86	−2·67
E_4	0	−0·45	−0·72	−0·72	−0·63	−0·42	0	+1·65	
$\int_0^{x/c}(u_1/U)^{1\cdot87}E_4\,d\left(\frac{x}{c}\right)$	0	−0·013	−0·10	−0·203	−0·340	−0·452	−0·508	−0·349	
$\frac{1}{(u_1/U)^{2\cdot87}}\int_0^{x/c}\left(\frac{u_1}{U}\right)^{1\cdot87}E_4\,d\left(\frac{x}{c}\right)$	0	−0·016	−0·050	−0·077	−0·114	−0·144	−0·158	−0·114	
$\frac{hc}{k}\Big/\left(\frac{Uc}{\nu}\right)$ (2nd approx.)	1·716	1·400	1·057	0·842	0·711	0·620	0·543	0·489	

No further cycles are needed

* From the front stagnation point relation.

point, equation (8) becomes

$$\left(\frac{\Delta_m}{c}\right)^2\left(\frac{Uc}{\nu}\right)=9\cdot89\Big/\frac{d(u_1/U)}{d(x/c)}.$$

$[d(u_1/U)/d(x/c)]$ is 12 at the front stagnation point of the 2:1 ellipse.

5. HEAT TRANSFER IN AXISYMMETRIC FLOW

By a transformation due to Mangler[11], we may alter the equations of motion and energy for axisymmetric flow into formal identity with the two-dimensional equations.

The new variables are:

$$\left.\begin{aligned}\bar{x}&=\int_0^x\frac{r^2}{c^2}\,dx\\ \bar{y}&=\frac{r}{c}\,.\,y\\ \bar{\Delta}&=\frac{r}{c}\,.\,\Delta\\ \bar{h}&=\frac{c}{r}\,.\,h\\ \bar{u}_1&=u_1\end{aligned}\right\}\quad\ldots\quad(11)$$

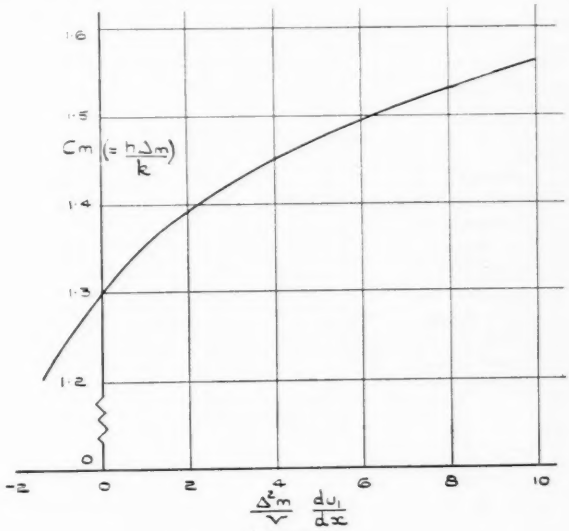

FIGURE 3.

TABLE II. Extract from calculations by Δ_m method.

x/C	0	0·1	0·2	0·3	0·4	0·5	0·6	0·7	0·8
u_1/U	0	0·926	1·273	1·400	1·461	1·491	1·500	1·495	1·466
$I=\int_0^{x/c}(u_1/U)\,d\left(\frac{x}{c}\right)$	0	0·0522	0·1649	0·2976	0·443	0·591	0·741	0·890	1·037
$\left(\frac{\Delta_m}{c}\right)^2\left(\frac{Uc}{\nu}\right)=\frac{19\cdot78\,I}{(u_1/U)^2}$	0·823*	1·203	2·014	3·002	4·11	5·26	6·52	7·82	9·54
$\left(\frac{\Delta_m}{c}\right)^2\left(\frac{Uc}{\nu}\right)\frac{d(x/c)}{d(u_1/U)}$	9·9	7·1	4·0	2·7	1·6	0·8	0·1	−1·25	−3·4
C_m from Fig. 3	1·56	1·52	1·45	1·41	1·37	1·34	1·31	1·23	
$\frac{hc}{k}\Big/\sqrt{\left(\frac{Uc}{\nu}\right)}=\frac{C_m}{\left(\frac{\Delta_m}{c}\right)\sqrt{\left(\frac{Uc}{\nu}\right)}}$	1·716	1·381	1·021	0·816	0·678	0·583	0·512	0·438	

*From the front stagnation point relation.

Given $u_1(x)$ and $r(x)$, $\overline{u}_1(\overline{x})$ may be tabulated and $\overline{\Delta}_m$ determined from equation (8) and hence $\overline{h}$ from (10), with C_m determined from Fig. 3, using $(\overline{\Delta}_m^2/\nu)\,d\overline{u}_1/dx$. $h(x)$ or the appropriate dimensionless form is then determined from $\overline{h}=(c/r)\,.\,h$.

APPENDIX I

The well-known "wedge" solutions (Eckert[1]) are for flows with $u_1 \propto x^m$ and constant wall temperature. Under these conditions both velocity and temperature layers are "similar," that is they do not alter in shape along the surface. The shapes and thickness of velocity and temperature layers are tabulated by Eckert. It is easy to show that the relation between the quantities in equation (1) and the dimensionless thicknesses in the wedge solutions is:

$$u_1 \frac{d}{dx}\left(\frac{\Delta_n^2}{\nu}\right) = 2Z_n^2\left(\frac{1-m}{1+m}\right)$$

$$\frac{\Delta_n^2}{\nu}\frac{du_1}{dx} = 2Z_n^2\left(\frac{m}{1+m}\right)$$

Eckert has tabulated Z_1, Z_2, Z_4 against $\beta=2m/1+m$, for various σ, and in consequence corresponding values of $(\Delta_n^2/\nu)\,u_1 d/dx$ and $(\Delta_n^2/\nu)\,du_1/dx$ are easily found, for wedge flows. These relations are plotted in Fig. 1 for Δ_1, Δ_2, Δ_4. The crucial assumption of the present paper is that these corresponding values hold also for flows other than wedge flows, that is for flows where u_1 is an arbitrary function of x.

APPENDIX II

Defining C_n by $(\partial\tau/\partial y)_0 = C_n/\Delta_n$, C_n is then dependent on the temperature layer shape for $n=1, 4$, and in addition on the velocity layer shape and thickness for $n=2$. Shape, by dimensional analysis similar to that leading to equation (1), is dependent on $(\Delta_n^2/\nu)\,du_1/dx$. For wedge flows Eckert has in effect tabulated $(\partial\tau/\partial y)_0\Delta_n$ as a function of $\beta=2m/1+m$ and hence as a function of $(\Delta_n^2/\nu)\,du_1/dx$. C_n is thus easily found as a function of $(\Delta_n^2/\nu)\,du_1/dx$. The values are tabulated for the "mixed" thickness Δ_m, for $\sigma=0{\cdot}7$.

β	1·0	0·5	0·2	0	−0·14
$\left(\frac{\Delta_m^2}{\nu}\right)\frac{du_1}{dx}$	9·89	4·88	1·96	0	−1·48
C_m	1·56	1·47	1·39	1·30	1·20
	↑ stagnation point			↑ flat plate	

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Approximate Deflections in Cantilevers Curved in Plan

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IT IS NOT DIFFICULT to establish equations by means of Castigliano's First Theorem intended to give the deflections and rotations at any point in a cantilever curved in plan. However, when the bending modulus, EI, and the torsional modulus, GJ, are unequal in magnitude and when the plan form is not of simple shape, these equations become difficult to handle, if not unmanageable. For the case in which $EI \simeq GJ$ (a solid circular section or annulus) it is possible to obtain simple, approximate, and explicit expressions for the deflections under a concentrated end load of W_1 and end couples M_o, T_o for any form of beam[1]. These expressions are now extended in Section 1 to take account of the end loading shown in Fig. 1. Section 2 treats of the deflections at a point other than the end at which the load is applied. The manner of doing this is the same as in Section 1 and explicit expressions are again yielded.

In the analysis, deflections due to shear force and direct stresses are neglected. The results obtained are approximate and their error varies but it is always less than 12 per cent.

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NOTATION

EI	bending modulus
GJ	torsional modulus
I_{zz} I_{yy}	second moments of the curve about B
I_{yz}	second product moments of the curve about B
J_B	polar moment of length of curve about B

M_c	bending couple at C
M_o, N_o, T_o	end couples at B
M_p, T_p	moments at P
(r, θ)	co-ordinates of P with respect to B
s	length of curve from A to B
T_c	twisting couple at C
U	strain energy due to bending and twisting
W_1, W_2	vertical loads
X, Y, Z	perpendicular forces acting at B
Δ_x	vertical deflection at B for beam AB
Δ_A	vertical downward deflection at A for beam AD
Δ_B	vertical downward deflection at B for beam AD
σ	Poisson's ratio
θ_{zz}	rotation or bending about axis BZ

1. Deflections under point of application of load

Figures 1(*a*) and (*b*) show, in plan, a plane horizontal member AB of uniform circular cross-section built-in at right angles to a wall at A. Three perpendicular forces X, Y, Z act at B, X being vertical and Y tangential to the beam at B, *see* Fig. 1(*a*). Also at B are three perpendicular moments M_o, N_o, T_o represented vectorially by the usual convention; N_o is vertical and T_o is tangential to the beam at B, *see* Fig. 1(*b*). AB is then subject to combined bending and torsion. The co-ordinates of P with respect to B and the tangent at B are (r, θ). If the couple T_p is parallel to the tangent at P and the couple M_p is in the plane of the cross-section at P then the resolved components of the end loads at B are as shown in Fig. 2. It can be shown that,

$$M_p^2 + T_p^2 = M_o^2 + T_o^2 + N_o^2 + X^2r^2 - 2M_oXr\cos\theta - 2T_oXr\sin\theta + Z^2r^2\cos^2\theta + Y^2r^2\sin^2\theta + 2N_oZr\cos\theta + 2N_oYr\sin\theta + 2ZYr^2\sin\theta\cos\theta \qquad (1)$$

Now applying Castigliano's First Theorem, U, the strain energy due to bending and twisting is,

$$U = \int_B^A \left(\frac{M_p^2}{2EI} + \frac{T_p^2}{2GJ}\right) ds \qquad (2)$$

For a beam which has a circular (solid or hollow) cross-section,

$$EI/GJ = (1+\sigma) \simeq 1{\cdot}3 \qquad (3)$$

Taking Poisson's ratio, $\sigma \simeq 0{\cdot}3$ for steel, from (2), using (3), it is found that,

$$2U(EI + GJ) = 2\int (M_p^2 + T_p^2)\, ds + 0{\cdot}3\int T_p^2 ds - 0{\cdot}23\int M_p^2 ds$$

or

$$U \simeq \frac{1}{EI + GJ}\int (M_p^2 + T_p^2)\, ds$$

$$\simeq \frac{4}{7EI}\int (M_p^2 + T_p^2)\, ds \qquad (4)$$

The vertical deflection at B is given by,

$$\Delta_x = \partial U/\partial X = \frac{4}{7EI}\frac{\partial}{\partial X}\left[\int (M_p^2 + T_p^2)\, ds\right] = \frac{8}{7EI}\left[XJ_B - s(M_o\bar{y} + T_o\bar{z})\right] \qquad (5a)$$

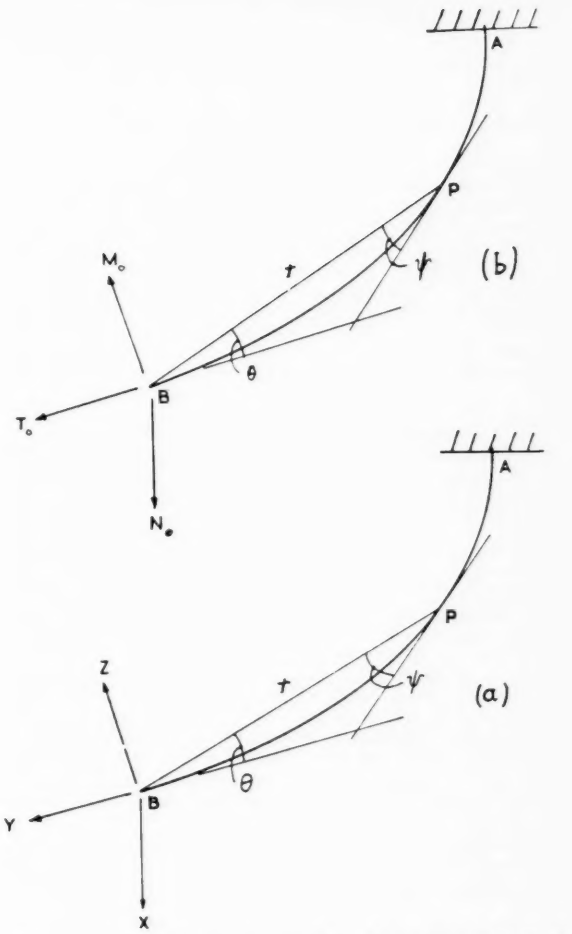


FIGURE 1. Cantilever beam curved in plan with end loads X, Y, Z and end couples M_o, N_o, T_o.

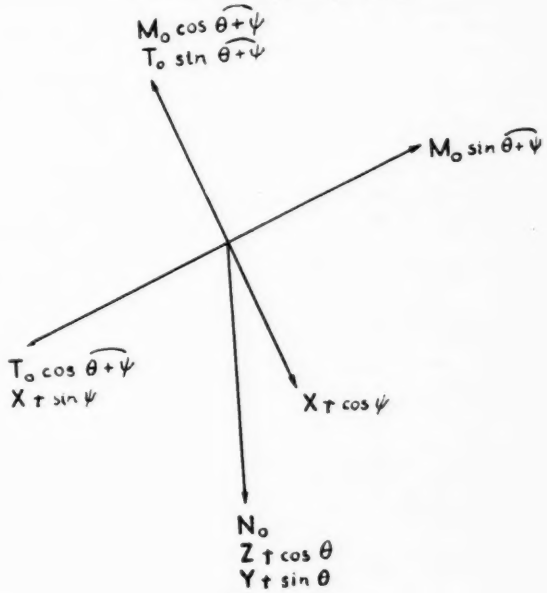


FIGURE 2.

using (1), in which

$$J_B = \int r^2 ds, \quad \bar{z} = \int r \cos\theta ds, \quad \bar{y} = \int r \sin\theta ds$$

and s is the length of the curve from B to A. Similarly,

$$\Delta_y = \frac{8}{7EI}\left[YI_{yy} + N_o s\bar{y} + ZI_{yz}\right] \quad . \quad (5b)$$

$$\Delta_z = \frac{8}{7EI}\left[ZI_{zz} + N_o s\bar{z} + YI_{zy}\right] . \quad . \quad (5c)$$

in which $I_{yy} = \int r^2 \sin^2\theta ds, \quad I_{zz} = \int r^2 \cos^2\theta ds$

and $I_{yz} = \int r^2 \sin\theta \cos\theta ds.$

$\bar{z}$, $\bar{y}$ in effect define the centroid of the beam, while I_{zz}, I_{yy} are the second moments of the curve about B; I_{yz} is the product second moment and J_B the polar moment of length of the curve about B. Similarly, it may be shown that θ_{zz} (rotation or bending about axis BZ),

$$= \frac{8s}{7EI}(M_o - X\bar{y}) \quad . \quad . \quad . \quad (5d)$$

$$\theta_{yy} = \frac{8s}{7EI}(T_o - X\bar{z}) \quad . \quad . \quad . \quad (5e)$$

$$\theta_{xx} = \frac{8s}{7EI}(N_o + Z\bar{y} + Y\bar{z}) \quad . \quad . \quad (5f)$$

2. Vertical deflection at any point of beam

In Fig. 3, the beam AD is loaded at A with two couples M_o, T_o and a vertical load W_1. A second vertical load W_2 acts at point B distant p from A.

2.1. *Vertical deflection at B*

At point C, the bending couple M_c and the twisting couple T_c are given by

$$M_c = M_o \cos(\theta_1 + \psi_1) + T_o \sin(\theta_1 + \psi_1) - W_1 r_1 \cos\psi_1 - W_2 r_2 \cos\psi_2 \quad (6a)$$

$$T_c = -M_o \sin(\theta_1 + \psi_1) + T_o \cos(\theta_1 + \psi_1) + W_1 r_1 \sin\psi_1 + W_2 r_2 \sin\psi_2 \quad (6b)$$

Now, $U = U_1(M_o, T_o, W_1) + U_2(M_o, T_o, W_1, W_2)$ where U_1 is the strain energy over length A to B and U_2 is the strain energy over length B to D.

The vertical downwards deflection at B is

$$\Delta_B = \frac{\partial U}{\partial W_2} = \frac{\partial U_2}{\partial W_2} = \frac{8}{7EI}\int_B^D \left(M_c \frac{\partial M_c}{\partial W_2} + T_c \frac{\partial T_c}{\partial W_2}\right) ds$$

and using (6) it is found that

$$\Delta_B = \frac{8}{7EI}\left[W_2\, {}_BJ_{BD} - (M_o \bar{x}_{BD} + T_o \bar{y}_{BD})\, s_{BD} + \frac{W_1}{2}({}_AJ_{BD} + {}_BJ_{BD} - p^2 s_{BD})\right] \quad (7)$$

where $${}_BJ_{BD} = \int_B^D r_2^2 ds, \quad r_2 \text{ being measured from } B.$$

$${}_AJ_{BD} = \int_B^C r_1^2 ds, \quad r_1 \text{ being measured from } A.$$

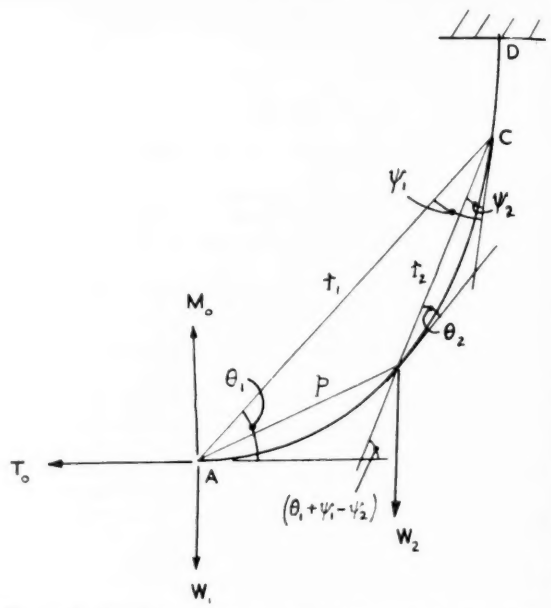


FIGURE 3. Cantilever beam curved in plan, carrying two vertical loads W_1, W_2 and end couples M_o, T_o.

$$s_{BD} \cdot {}_B\bar{y}_{BD} = \int_B^D r_2 \sin(\theta_1 + \psi_1 - \psi_2)\, ds$$

and $$s_{BD} \cdot {}_B\bar{x}_{BD} = \int_B^D r_2 \cos(\theta_1 + \psi_1 - \psi_2)\, ds.$$

If $M_o = T_o = 0$,

$$\Delta_B = \frac{8}{7EI}\left[W_2\, {}_BJ_{BD} + \frac{W_1}{2}({}_AJ_{BD} + {}_BJ_{BD} - p^2 s_{BD})\right] \quad (8)$$

2.2. *Vertical deflection at A*

In this instance,

$$\Delta_A = \frac{\partial U}{\partial W_1} = \frac{\partial U_1}{\partial W_1} + \frac{\partial U_2}{\partial W_1} \quad . \quad . \quad . \quad . \quad (9)$$

Proceeding as in 2.1 and simplifying, it is found that

$$\frac{\partial U_1}{\partial W_1} = \frac{8}{7EI}\left[W_1\, {}_AJ_{AB} - s_{AB}(M_o\, {}_A\bar{x}_{AB} + T_o\, {}_A\bar{y}_{AB})\right]. \quad (10a)$$

$$\frac{\partial U_2}{\partial W_1} = \frac{8}{7EI}\left[W_1\, {}_AJ_{BD} - s_{BD}(M_o\, {}_A\bar{x}_{BD} - T_o\, {}_A\bar{y}_{BC}) + \frac{W_2}{2}({}_AJ_{BD} + {}_BJ_{BD} - p^2 s_{BD})\right]. \quad (10b)$$

Thus using (9) and (10) Δ_A is obtainable.

In particular when $M_o = T_o = 0$, then

$$\Delta_A = \frac{8}{7EI}\left[W_1({}_AJ_{AB} + {}_AJ_{BD}) + \frac{W_2}{2}({}_AJ_{BD} + {}_BJ_{BD} - p^2 s_{BD})\right]$$

REFERENCE

1. JOHNSON, W. (1955). Expressions for Circular Section Bow Girders Carrying a Concentrated Load. *The Engineer*, p. 145. July 1955.

Safety and Large Aircraft

Comment by J. LAURENCE PRITCHARD, *Hon. Fellow*

MAY I repeat the first paragraph of Mr. A. H. Crawshaw's letter in the December 1957 Journal?

"Our Journal is becoming less and less interesting. Anyone who doubts this should look back at some of the old smaller Journals. It (*sic*) contains more and more mathematics and less and less engineering. The proportion of recipients who read it must be diminishing and this surely defeats its object."

Having edited "the old smaller Journals" for half their total lives, may I say that, like *Punch*, any particular number is never as good as it was? The Journal is just as good as the members like to make it. If they choose not to read it, then it is not worth reading, *from their point of view*. I should advise, nevertheless, that such members keep their Journals carefully, for they fetch a good price in the open market.

I blame members of the Society, without reserve, for any attitude of complacency towards their Journal; towards the papers read before them; towards any activity of the Society with which they are in disagreement. Too many members expect to be spoon fed, and if they cannot stomach the medicine in the spoon they cannot deny they have asked for it. They are not playing their proper part in the Society, for they owe far more to it than just their subscriptions.

The Society has a Charter of Incorporation and every member has received a copy of it. I strongly recommend those who have not read it to do so, and particularly those paragraphs which declare, *under the Charter* "the objects for which the Society is hereby constituted."

Here I need only quote "4a. To facilitate the exchange of information and ideas *amongst the members of the Society*." (My italics.) That means among *all* members, so don't be deluded by that specious word *restricted*. The Society does not exist for, nor is concerned with, *under its Charter*, information which is not open to all members. The members are either all goats or all sheep, but not a mixture. The Society is not a secret Society. If you lose freedom of open discussion you lose your freedom indeed.

May I quote Mr. Crawshaw again?

"I believe it was a grave error of the aircraft industry to accept a factor of two only as being sufficient for a pressure cabin when a factor of 5 or 6 was considered necessary for a boiler."

In a work on Aeroplane Structures published in 1919, the authors wrote, with regard to aeronautical engineering:

"If an aeroplane has under any conditions of flight to carry N times its normal steady flight load, the structure must be designed accordingly, and is said to have a load factor of N.

"The load factor is often spoken of as the factor of safety of the aeroplane, but this is an error which should be carefully avoided. Factor of safety is a term which has a perfectly definite meaning well understood in other branches of engineering, and it should be reserved for that purpose. When a structure such as a bridge is designed, the worst possible loads which can come upon it are estimated, and the bridge is then designed so that when this maximum load is carried the material is only stressed to about one fifth of its ultimate. In other words the bridge could carry five times the greatest possible load it ever has to bear before failure would occur. This figure of 5 is then the factor of safety of the structure.

"With an aeroplane, however, the case is different. The least possible load to maintain flight in a straight line is estimated, and factors are based on this. A small scout plane may well have a load factor of seven but a factor of safety of under one; that is to say, it would be possible by clumsy flying to break the aeroplane in the air.

"If the worst load occurring under any flight conditions could be estimated, it might be possible to estimate a true factor of safety for an aeroplane, but at the present moment this is not customary, at any rate for the wings."

Can it be said that there is any fundamental change from the days of 40 years ago about the meaning of the words "factor of safety" in the aeronautical sense? Or does the phrase "factor of safety" now join with "clean bomb" of bitter modernity?

I have referred to a book on Aeroplane Structures. In the Preface the authors wrote: "It is recognised by the Authors that this branch of engineering is an entirely new one, and much criticism of the first book devoted to aeroplane structures is to be expected. Such criticism will be welcomed, as it is only by the mutual exchange of ideas that progress in a new science can be made."

As one who believes the Journal should be a platform for controversial aircraft matters I hope Mr. Crawshaw has started something in time to save the Society altering the title to the "Missile Magazine and Rocket Record."

Comment by R. K. PAGE, *Associate Fellow*,
Chief Technician (Projects) Folland Aircraft Ltd.

AIRCRAFT engineering needs more and more mathematics and so inevitably a Journal keeping up to date with the subject becomes less easy reading to its older readers. The total membership of the R.Ae.S. is increasing and there is no reason to suppose that an equally large proportion of the newer and younger members do not find the aircraft design field as exciting as we did 20 years ago.

Superficially it appears reasonable to say that as the direct material and personal consequences of a failure become greater, so should the attention paid to avoiding that failure be increased, but the implied converse in the present application that the structural safety of a small transport aircraft is relatively unimportant would surely be rejected by operators and authorities alike. The blind use of high factors of safety which have really been very approximate "factors of ignorance" is not the way to get a safe aircraft structure that is at the same time reasonably light in weight, which latter is of course essential not only for economic reasons, but to avoid defeating the object by reducing the performance in those aspects, such as rate of climb that also directly bear on safety.

A prime difference between the aircraft designer's factor of 2 on *his* pressure vessel and the boiler engineer's factor of 5 or 6 is that the former designs on the most adverse conditions as far as he is able to estimate them, and therefore needs only a relatively small further margin, whereas the latter usually designs on normal operating conditions and material strengths and allows for such adverse deviations as corrosion and occasional excessively severe conditions of load and temperature (possibly due to mishandling), by including their effects in a large factor of safety.

It is surely rather a naïve trust in the power of officialdom to believe that the safety of any engineering structure can be increased by decree! On the figures quoted, an immediate *doubling* of the factor, *e.g.* by halving the pressure differential after the first accident would probably not have prevented a second. What *will* produce safer aircraft, is the process that has in fact evolved, namely, an attempt to find out what *is* the safe stress for a particular material in particular conditions (which includes its incorporation by practical fabrication processes into a realistic structure), and the subsidiary problem of finding out how to avoid locally high concentrations of stress that will initiate major failure.

"Fail Safe," like "Safety First," is an excellent principle applied in moderation and with commonsense, but followed blindly to its logical conclusion would lead to such absurdities as all civil transports being biplanes. One has heard before of the helicopter that *could* continue flying minus one of its blades, but not how it is proposed to ensure that in the instant after its "safe failure" the superfluous blade does not slice open the cabin or remove the tail rotor.

Graduates' and Students' Section

Lecture

"Trends in Air Transport" was the subject taken by Mr. P. W. Brooks, Technical Assistant to the Chairman of B.E.A., in his lecture on 16th October. His talk covered a subject of wide interest to all in the aircraft industry now that it must look more to the civil field for future work.

Mr. Brooks reviewed the expansion of civil aviation as an industry, and indicated the steady improvement in economy being achieved; the comparison between pre-war and current types being particularly outstanding. From this point it was shown how a logical development, such as physical stretch or engine development, can be used to improve economy.

Some ideas as to the future trends which the U.K. industry ought to follow were then evolved, using as an indication the comparative histories of civil aircraft development in both Britain and the U.S.A. In particular, by reference to the "Brabazon Committee" types, and the sorry financial history of some of those aircraft, the lecturer was able to put forward a strong case for trying to develop at the same rate as the Americans, and not to try to leap-frog ahead of them.

Finally, Mr. Brooks gave some of his impressions as to the future demands for aircraft of the high subsonic speed régime, and thence to the supersonic régime, with all its attendant complications.

The questions which followed were many and varied, and an interesting argument developed between protagonists of the turbo-prop and the pure jet. The volume of questions was an indication of the high standard of the lecture, and the interest it provoked among the audience.

Annual Dance

This year's annual dance was held on the last Friday in November. With dancing to Hugh McCamley and his band, it was enjoyed by over a hundred graduates, students, wives and girl friends. It was nice to see members from as far away as Southampton.

There was wine, red and white, for the asking as well as potent-but-popular vintage cider. For the more sober minded there were sandwiches and coffee as well. In the novelty numbers many and exotic were the versions seen of rock 'n' roll and dances from foreign lands.

Our photographs show the scene in the library of No. 4 Hamilton Place when festivities were at their height.

Lecture—28th January

The Section's first lecture in 1958 will be given on 28th January by Mr. K. G. Wilkinson on "Developments in Glider Design"—in the Library, 4 Hamilton Place, at 7.30 p.m.

Visit

Wet weather failed to daunt 25 members of the Section who visited the Road Research Laboratory at Langley last month and it proved of great interest, especially to motoring enthusiasts.

In the Slipperiness Department we saw the effect of the various surface finishes on skid resistance, and of the degrees of wetness usually encountered. Obviously the best possible brakes will not have the required effect if the coefficient of friction between the tyre and the surface is too low. This adhesion was seen to depend on speed, generally diminishing as speeds increase, with obvious consequent difficulties in the case of the braking of aircraft at large velocities. A mobile slipperiness testing machine, affectionately known as "Marmaduke," was demonstrated and the excellent, although unfortunate, lubricating properties of water on rubber were emphasised.

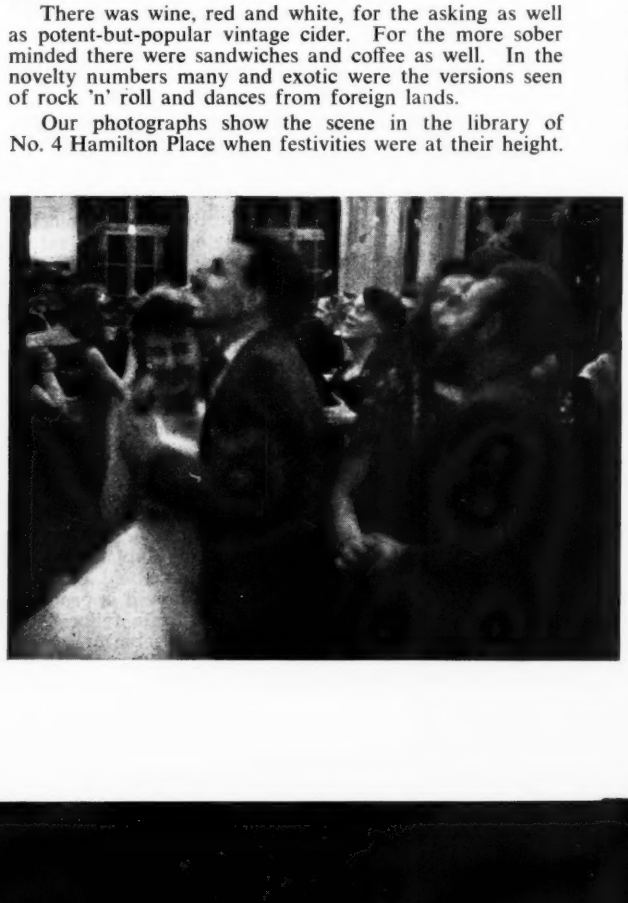
In the lighting section a variety of vehicle headlights and spotlights were mounted for comparative testing of their efficiencies. The relative merits of such properties as sharp cut off across the top of the beam and a variety of dipping devices were explained to us.

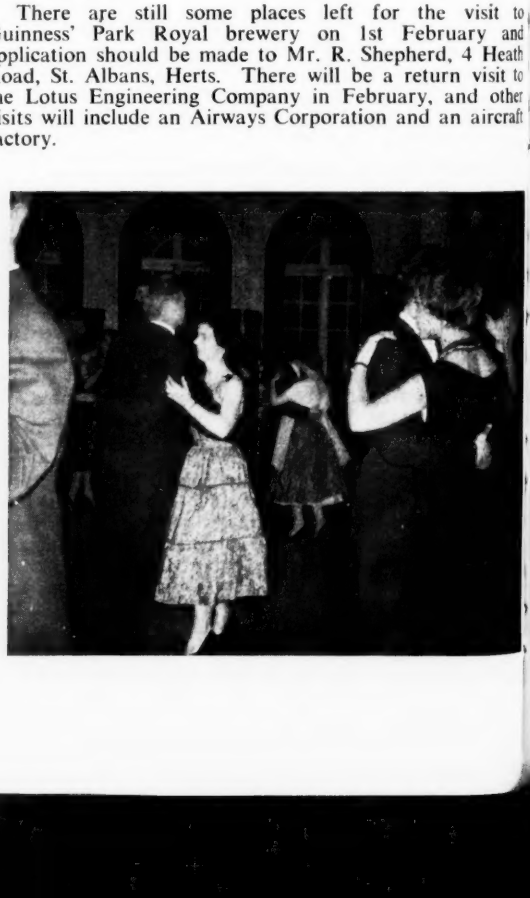
The vehicle inspection department was visited last. The work there covered the investigation of testing standards for brakes, lights and steering, and of crash injury analysis. The startling discovery that more people are injured when travelling in (or on) a vehicle than as pedestrians increased our respect for the research on such aids as safety belts. Perhaps there is scope for the motor industry to introduce rearward facing seats!

We were reassured to see that the R.R.L. adopts a scientific and realistic approach to our national road problems.

Future Visits

There are still some places left for the visit to Guinness' Park Royal brewery on 1st February and application should be made to Mr. R. Shepherd, 4 Heath Road, St. Albans, Herts. There will be a return visit to the Lotus Engineering Company in February, and other visits will include an Airways Corporation and an aircraft factory.





THE LIBRARY

Reviews

A TREATISE ON PHOTO-ELASTICITY. *E. G. Coker and L. N. G. Filon, revised by H. T. Jessop. Cambridge University Press, 1957. 2nd edition. 720 pp. Illustrated. 70s.*

Photoelasticity is a particularly attractive branch of the science of stress analysis, and many will be familiar with the monochromatic fringes or the picturesque coloured isochromatics which can be seen in models and which enable stresses to be evaluated. Sir David Brewster discovered the doubly-refracting properties of stressed transparent materials in 1815, and the theory was effectively completed with Maxwell's enunciation of the stress-optical laws of behaviour in 1850. However, the potentialities of the method only became widely appreciated with the publication of Coker and Filon's remarkable and varied contributions from 1910 onwards, culminating with their classic book on the subject in 1931. The first edition laid the foundations of fundamental knowledge on photoelasticity, and so helped to build the subject to its present refined and versatile stage of development. The new edition is welcomed, not only because the first has been out of print for some years, but because a new introduction, giving a brief though excellent review of the developments that have occurred since the first publication, has been added by Colonel Jessop, one-time associate of the original authors.

The main text is precisely the same as the original, except that the exceedingly few errors have been eliminated. The behaviour of light in stressed models is described rigorously in mathematical terms, and the same presentation is justified as no improved theories have been introduced. The mathematics is of a high standard, and the book can hardly be used lightly for purposes of reference, but rather for a thorough perusal. Perhaps this explains the absence of a list of notations, although, curiously, mention is made of the "diversity of notation employed by mathematical and engineering writers." None of the new techniques which are so important in modern photoelasticity are described in detail, and yet the behaviour and analysis could scarcely be deduced from the information given. Furthermore, the stress values ascertained from models would not be considered reliable by modern standards.

To sum up, the book forms a fitting symbol of the splendid pioneer work of the authors, and is still a useful ultimate source of reference to the fundamental behaviour of light in stressed models. It should be of value to those engaged in pure research on the subject, but to those applying the method for the solution of modern stress problems the book would give only an inkling of developments and techniques that are now available.—R. B. HEYWOOD.

GUIDED WEAPONS. *Eric Burgess. Chapman & Hall, London, 1957. 225 pp. Illustrated. 25s.*

A major part of the book is devoted to descriptions of German and American missile developments. British security restrictions have obviously been a grave handicap, but it is gratifying to see a sequence of photographs showing the destruction of a target drone by the British Fairey Fireflash air-to-air missile.

The chapter on Propulsion gives much detailed information, but the presentation is disorderly, many of the factual data are erroneous and the basic principles are not always clear. Tables of propellant performance especially contain numerous incorrect values, and the division of solid propellants into unrestricted and restricted burning types has little significance. In the chapter on Guidance and Control the author mixes a series of generalities with descriptions of particular guidance components, followed by sections on instrumentation which would be better included in the chapter on Testing.

Throughout the book there is a wealth of photographs, mostly of American missile firings, including several interesting records of American experimental ballistic and high altitude missiles; in particular there is an early firing of the Convair MX774 which is said to be the forerunner of the American Atlas 5,000 miles I.C.B.M. The general impression is, however, one of recurring intrusion of irrelevant details, both in the text and the illustrations. There is not really much which can be recommended in the book, either for the general reader, for whom it contains too much haphazard detail, or for the technical man, for whom the errors seriously restrict its value.—W. H. STEPHENS.

MINIMUM WEIGHT ANALYSIS OF COMPRESSION STRUCTURES. *George Gerard. University Press, New York, 1956. 194 pp. Illustrated. $6.00.*

Prophecy is dangerous, particularly in the field of engineering. One of the most consistent prophecies in aeronautics has been "thus far and no larger," and the Square Cube Law has been much bandied about. It is to the greatest credit of the aircraft engineering profession that this "Law" has continually been broken by the production of ever more efficient structures. In recent times one of the most important weapons in the battle against the penalties of size has been the scientific analysis of structural efficiency. Dr. Gerard has produced a reference book covering part of the field of this science. He has confined himself to compression structures, and has dealt almost entirely with individual components, leaving the project designer to make the overall choice of structural disposition.

He begins by considering the efficiency of the simple structural elements: columns and plates and their combination in thin-walled columns. This chapter is crucial to all which follows, since it contains basically all the techniques used in succeeding chapters. Yet it is written in a throw-away style, and the basic principle of simultaneous buckling is treated very casually. The various non-dimensional expressions appear like rabbits from a conjuror's hat; and the chapter would become much more useful to the reader unfamiliar with the field if it contained more explanation as to the purpose of the processes noted.

Having dealt with the foundations, the author follows with detailed consideration of the various typical composite structural components and produces the necessary criteria for minimising the weight of such items as stringer-skin, sandwich plate, multi-cell and post construction of wings. There is also a chapter on cylinders. The final third of the book covers the synthesis of what has gone before. A chapter on comparative efficiency in bending wisely reminds the reader that structures primarily designed

by bending considerations also carry shear and have to meet stiffness requirements. This is followed by very sensible accounts of the practical limitations, the relative efficiency of materials, and the ideal stress for minimum weight. The book contains copious references, a review of relevant literature, a full bibliography, and an adequate index.

This volume is more truly a reference book than a text book, and so the remarks made about the initial chapter may seem out of place. However, the excellence of the reference book which Dr. Gerard has written does underline the need for a work which has a wider canvas; which concerns itself more fully with the aims and ends of minimum-weight analysis as well as with the techniques. The successes of such analysis are only achieved when they are transformed into engineering structures, and so much of what is gained by the use of the methods Dr. Gerard has outlined is thrown away by careless detailing and by lack of concern about secondary structure. It is sincerely to be hoped that Dr. Gerard will continue his good work by giving us a true introduction to this subject which is of such vital importance to the aircraft industry.—K. H. GRIFFIN.

THE SPECTROSCOPY OF FLAMES. *Dr. A. G. Gaydon. Chapman & Hall, London. 1957. 279 pp. Illustrated. 50s.*

This authoritative monograph effectively replaces Dr. Gaydon's earlier book *Spectroscopy and Combustion Theory* (2nd edn. 1948); it brings the subject matter up-to-date but covers somewhat different ground, material in the author's books *Dissociation Energies* and (with Dr. Wolfhard) *Flames, their Structure, Radiation and Temperature* published in 1953 having been omitted and some new material having been introduced. The new book is some 40 per cent larger, containing 279 pages of 20 per cent greater content, as compared with the 242 pages of the earlier book. There is no doubt in the reviewer's mind that the latter book stimulated much research in the field of flame spectroscopy, just as Dr. Gaydon hoped it would. The frequent references to that book which have been made during every combustion symposium and colloquium during the past eight years testify to this, as also does the presence of nearly 200 additional references in the book under review.

The past decade has seen the introduction of a number of novel experimental techniques for controlling flames or for exciting spectra in special sources. There include flat flame and low-pressure flame burners, flash photolysis, shock-tube excitation and flames supported by free atoms, many of which, having been developed at Imperial College by Dr. Gaydon and his colleagues, are described from first hand knowledge. Other new material in the book includes a discussion of flames supported by halogens and by oxides of nitrogen, and a new chapter in which the application to flame spectrophotometry of knowledge of flame structure and excitation conditions in flames is discussed.

As in the earlier book, chapters are devoted to hydrogen flames, carbon monoxide flames, and by hydrocarbon flames (now called organic flames), to the infra-red region, to the measurement of flame temperature and to the nature and significance of engine explosion spectra. In every case, however, the earlier treatment has been most carefully revised, expanded where necessary and brought up-to-date.

In a most interesting and stimulating chapter entitled Flame Structure and Reaction Processes, Dr. Gaydon expresses his personal opinions on the interpretations of numerous flame spectra and their bearing on the theories of chemical reaction processes (*e.g.* cool flame reactions, carbon formation in flames, hydrocarbon oxidation mechanisms) and on the detailed structure of the reaction zone.

Many quantitative data on wavelengths of band spectra and absorption spectra and a summary of the main features of premixed-, decomposition- and atomic-flame emission spectra are included in Appendices.

This very readable monograph will prove to be indispensable to many research workers in the combustion field. Although having much in common with *Spectroscopy and Combustion Theory*, this is essentially a new book and replaces the earlier one as the standard work on a subject pioneered by Dr. Gaydon who is, in addition, the leading authority.

In view of the greater size of the book, the better quality of the paper and the binding, and the intervening period of inflation, the doubling of the price as compared with *Spectroscopy and Combustion Theory* is regarded as reasonable.—B. P. MULLINS.

NUMERICAL METHODS. *R. A. Buckingham. Pitman, London, 1957. 597 pp. 70s.*

With the present growth of the subject, there are broadly speaking, two ways in which a book on numerical analysis may be written. The first, and in the opinion of the reviewer the more desirable way, is to select a certain number of topics and to treat them as thoroughly as possible. The alternative is to try to deal with all the aspects of the subject in one volume, to produce a compendium as it were. Because of the limitations of space, the second approach does not permit the material to be treated adequately, this frequently being noticed by the omission of remarks on the treatment of difficulties which will arise in a blind, inexperienced application of the methods described.

The volume by Dr. Buckingham falls into the second classification and so naturally suffers from the drawbacks mentioned. The author has indeed minimised these objections by concentrating on the *methods* of numerical analysis but, even so, possible difficulties are not adequately treated and, by the omission of the background of numerical analysis, the serious student who is interested in more than a mere application of numerical methods will find little in the volume not covered by other treatments of the subject. To the user, however, it will appear as an interesting, instructive, and readable account of numerical methods.

An introductory chapter deals with such matters as errors, tables, and mechanical equipment available. Then follow a succession of chapters devoted to polynomial approximation, beginning with sums and difference of polynomials and properties of factorial polynomials. Lagrange methods are next treated, followed by methods using differences at unequal and equal intervals. There is a chapter devoted to the use of symbolic methods.

A brief treatment of step-by-step methods for the solution of ordinary differential equations follows, and particular attention is given to various approximate and partly analytic methods which have been devised for the solution of types of equations frequently occurring in physical problems. This is succeeded by a chapter on the solution of algebraic and polynomial equations and one on the method of least squares.

Matrices and allied problems, including the solution of linear simultaneous equations and the determination of eigenvalues, are covered in the next three chapters, this leading to the solution of ordinary differential equations with two-point boundary conditions and also those

involving latent roots. Integral equations of the Volterra and Fredholm types are also considered.

Final chapters deal briefly with two-dimensional work, including interpolation, numerical differentiation and integration, and the solution of simple partial differential equations.—D. C. GILLES.

GAS TURBINE MATERIALS. *G. Lucas and J. F. Pollock. Temple Press, London, 1957. 163 pp. Illustrated. 25s. net.*

This book presents an interesting and informative survey of past and present activities in developing materials for use in gas turbines. In dealing with this complex and ever-expanding field, it is evident that the authors did not intend a comprehensive treatment of the subject such as would satisfy the requirements of the pure metallurgist and engine designer, but rather their aim is to bridge the gap between metallurgy and design and performance technology. To this end, they have exercised great diligence, with the result that the work should prove to be of much value to students, engineers and metallurgists seeking a reliable and up-to-date review of gas turbine materials, and an understanding of the design philosophy governing the selection of a particular material for a specific application.

General impressions from a first reading of the book are:—

(*a*) The presentation of photographs of various gas turbine engines seems to contribute little to the development of the subject matter and, in the reviewer's opinion, diagrammatical layouts of hypothetical engines, showing temperature ranges for the main components together with recommended materials would be more instructive.

(*b*) There are notable omissions, *e.g.* sealing gland materials, and bearing materials and lubricants for high temperature service.

The extent of the authors' treatment is best indicated by the following abridged list of contents.

In the first chapter "An Outline of the Development of High Temperature Materials" the historical references to the early work on the isolation of the metallic elements such as chromium by Vauquelin (France 1798), and so on, seem almost irrelevant to the subject of "gas turbine materials," which is hardly two decades old, although this somewhat academic textbook approach might have a certain appeal to chemistry and metallurgical students.

Again, when considering "Early Ferritic Gas Turbine Steels" of the 3 per cent-Cr-Mo-V-W type, the sole reference to the Jessop-Saville (H.40) version ignores the major contributions of other authorities, particularly on such aspects as the metallurgical significance of developing a coarse bainitic structure by heat treatment, and the importance of vanadium-carbon ratio for optimum creep resistance.

Chapter 2 "The Influence of Performance Requirements on Operating Conditions of the Gas Turbine" discusses the operating conditions, with special reference to materials selection and to fuels. In the reviewer's opinion the damping capacity of the material in minimising fatigue failures is generally insignificant compared with the vibrational energy dissipated in the blade fixation. It would appear, therefore, that the prevention of fatigue failures depends in the main on designing a blade with characteristic modes of vibration which are not coincident with the spectrum of significant excitation frequencies found in the running range; with "fixation" methods which provide good damping characteristics; and with blade geometry and material which offers high fatigue efficiency.

"Metals under Stress and Temperature" is the subject of Chapter 3. In view of the number of fatigue failures of discs and blades which do occur, the reviewer considers that the treatment of the fatigue behaviour of metals has been somewhat cursory.

Figure 22, page 65, shows the presentation of creep data; but linearity of fracture and strain relationships is seldom achieved unless stress and time are on a logarithmic scale.

Chapters 4, 5 and 6 deal respectively with "The Application of High-Temperature Materials," "Special High-Temperature Materials" and "The Manufacture of Gas Turbine Components."

In conclusion both the authors and the publisher are to be congratulated on an excellent production which can be wholeheartedly recommended to anyone even remotely interested in this important subject.—W. J. HARRIS.

BRITISH AEROPLANES 1914-1918. *J. M. Bruce. Putnam, London, 1957. 742 pp. Illustrated. 12 gns.*

Comparisons may be odious, but with this book they are useless. No book has ever been published which describes and illustrates the British Aeroplanes of the 1914-1918 War so well as J. M. Bruce's new book. In that brief period of four years, under the lash of war, progress in the design of aircraft was made at a rate which was greater than ever before or has been since.

J. M. Bruce has written little short of half a million words and has provided 657 illustrations for this beautiful book, many of them from private collections and the majority of these previously unpublished. The book includes detailed notes on the operational development of each basic type, with many interesting items on modifications which were incorporated from time to time. Bruce has also provided an astonishing amount of information on particular machines and the units to which they belonged, with specifications and detailed performance flying.

The research which he has made to be able to describe all known types of British Aircraft, many of them obscure and some almost unknown, which gave service or were designed in the 1914-1918 War is awesome. He does not claim, recklessly, to include every British aeroplane of the period but he states in his Notes on the Use of the Book ". . . this history contains reference to all types and variants known to the Author." Knowing the standard that Bruce has set himself in all his writings on this subject there can be no doubt of the authenticity of this work.

A point or two of criticism might be added without being unfair. It is a pity that the Chief Designers' names are omitted from so many of the types described. This information is always useful to those who are engaged in historical research and who are most likely to use the book. Twelve guineas, alas, will put it out of reach of nearly all collectors and it will probably be found only in libraries, where it is unlikely to be available on loan. This does not detract from the value of the book, of course, and it will remain a standard work of reference unless and until some one writes a better one, which is improbable.—A.S.C.L.

SOLUTION OF PROBLEMS IN STRENGTH OF MATERIALS. *S. A. Urry. Second Edition. Pitman, London, 1957. 406 pp. Diagrams. 20s.*

The previous edition of this book was published in 1953 and reviewed in this Journal for September of that year (p. 605), where it was described as "a very useful book providing a wide range of fully worked examples

together with further practice examples with answers." In this second edition, the author has added several new problems and test papers and a new chapter on "Mechanical Properties and Theories of Failure."

SURVIVAL IN THE SKY. *Charles Coombs. Robert Hale Ltd., London. 124 pp. Illustrated. 15s.*

Mr. Coombs' book "Survival in the Sky" is based on three months association with aeronautical research centres in the U.S.A. The book is principally concerned with the physiological problems of flight and, where he keeps to generalities, Mr. Coombs remains on the right track. Unfortunately, he often departs from this to describe physiological mechanisms. On a number of occasions his descriptions are both inaccurate and misleading. For example: "red out . . . is caused by negative gravity which forces the heavily veined lower eyelids up over the eyeball. Looking through this veil of tissue gives the flyer the effect of 'seeing red'." (p. 50). On page 97, referring to the disorientating effects of high angular accelerations he writes "Under the . . . conditions of high speed flight, the liquid *surrounding* the semi-circular canals fizzes up like root beer when you shake it." (Reviewer's italics.—Ed.)

If in its proof stage it had passed through the hands of experts for checking, it would have been a most useful little book giving, in non-specialist language, an overall picture of the difficulties involved in putting man into a high performance flying machine. Because it is interesting and easy to read it may have popular appeal, but there is no doubt that it will also give the unsuspecting reader some strange and unusual ideas on the subject of human physiology.—T. C. D. WHITESIDE.

MODERN COMPUTING METHODS. *National Physical Laboratory. H.M.S.O., London, 1957. 128 pp. 10s. 6d.*

The advent of high-speed digital computers is having far-reaching effects in mathematics, physics and engineering. Many problems which up to now were intractable because of their size and complexity, can now be solved by numerical methods. The importance of a good understanding of computing methods has been correspondingly enhanced, for a poor method will at best waste valuable machine time and at worst give an incorrect result.

Modern Computing Methods gives a brief account of those computing methods likely to be of general use. Its emphasis is essentially practical and simple worked examples are included. There are chapters on algebraic equations and matrices, finite difference methods, ordinary and partial differential equations, and so on. There is an extensive bibliography.

Additions to the Library

Aerodynamic Forces on an Aerofoil in Non-Uniform Unsteady Motion in a Closed Tunnel, The. Rosenblat, S. *Phil. Trans. A977 Royal Society.* 10s. 1957.

"Aeroplane" Pictorial Review No. 2. Temple Press. 10s. 6d. 1957. Number 2 of a series of collections of photographs illustrating contemporary trends.

Air Cargo—The Next Ten Years. Brewer, S. H. Boeing Airplane Co. 1957.

***A.R.C. Technical Report of the Aeronautical Research Council,** Special volumes I, II, III. H.M.S.O. 22 gns. 1955-57. A collection of Reports and Memoranda that have been "left over" from the Annual Reports due to wartime restrictions, and so on.

***Aviation Dictionary. German/English: English/German.** Cescotti, R. (Second Edition.) Ward Lock. 15s. 1957. Originally published in Germany (by Hanns Reich Verlag, Munich) in 1954 and Ward Lock are to be congratulated on appreciating its value to an English market.

Discussion on Work-Hardening and Fatigue in Metals, A. Mott, N. F. (*Leader*). *Proc. A1229 Royal Society.* 18s. 1957. Twelve papers by twenty different authorities.

Elements of Heat Transfer. Jakob, M. and Hawkins, G. A. Third Edition. Wiley. 54s. 1957. *To be reviewed.*

Guide to the Aircraft Maintenance Engineers' Licence Examinations, A. Society of Licensed Aircraft Engineers. S.L.A.E. 10s. 1957. The fourth edition of the guide which has not been revised since 1949.

Handbook of Acoustic Noise Control. Bolt, R. H. *et al.* Vol. I. **Physical Acoustics.** W.A.D.C. 36s. 1952.

Handbook of Acoustic Noise Control. Vol. I. **Physical Acoustics. Supplement 1.** Lukasik, S. J. and Nolle, A. W. (*Editors*). W.A.D.C. 27s. 1955.

Handbook of Acoustic Noise Control. Vol. II. **Noise and Man.** Rosenblith, W. A. and Stevens, K. N. W.A.D.C. 72s. 1953.

Handbook of Noise Control. Harris, Cyril M. McGraw-Hill. 128s. 1957. *To be reviewed.*

Heat Transfer and Fluid Mechanics Institute, 1957. Cal. Inst. of Technology. Preprints. Stanford (O.U.P.). 68s. 1957. A collection of 21 papers (without discussion) presented at the 1957 Heat Transfer and Fluid Mechanics Institute. Many of them, all by Canadian or American authors, will appear later in the Journals of the sponsoring professional societies. This paper-covered volume has been produced photographically from the original manuscripts.

Hydraulic Surge in Airfield Hydrant Systems. Williams, P. G. B.P. Co. Ltd. 1957. Investigates a difficulty arising from fuelling by underground pipeline or hydrant systems in conjunction with automatic shut-off valves.

Improved Solutions of the Falkner and Skan Boundary Layer Equation. Smith, A. M. O. (S.M.F. Fund Paper FF-10). Inst. Aero. Sci. 75 cents. 1954.

Jet Aircraft Power Systems. Casamassa, J. V. and Bent, R. D. Second Edition. McGraw-Hill. 60s. 1957. First reviewed in the April 1951 Journal (p. 260). Seventy per cent of this second edition is new material. A "condensate of available information" on the "basics" of aircraft powerplants," including rockets, ram-jets, pulse-jets, turbo-jets and turbo-props, it is intended for students and is based on American engines.

Magnetohydrodynamics: A Symposium. Landshoff, R. K. M. (*Editor*). Stanford (O.U.P.). 32s. 1957. *To be reviewed.*

On the Minimum Weight of Certain Redundant Structures. Barta, J. Reprint from *Acta Technica Ac. Sc. Hungaricae.* 1957.

Pilgrims and Pioneers. Brittain, *Sir* H. Hutchinson. 1947. An old book only recently acquired.

Report of the London Airport Development Committee. M.T.C.A. C.A.P.145. H.M.S.O. 2s. 1957.

Strength of Materials. Shanley, F. R. McGraw-Hill. 64s. 1957. *To be reviewed.*

Table of Coefficients for Obtaining the Second Derivative without Differences. Salzer, H. E. and Robertson, P. T. Convair. 1957.

Theorie schallnaher Stromungen. Guderley, K. G. Springer. 42 DM. 1957. *To be reviewed.*

***World Aviation Directory, Fall-Winter 1957-8.** Gramlow, M. E. (*Editor*). American Aviation Publications. 10 dollars. 1957.

* For reference only.

Reports

AERODYNAMICS

BOUNDARY LAYER—*see also* THERMO-AERODYNAMICS TESTING AND INSTRUMENTS

Calculations of the pressure distributions and boundary-layer development on a body of revolution with various parabolic afterbodies at supersonic speeds. L. E. Fraenkel. R. & M. 2966. 1957.

Detailed calculations are made of the flow over a series of bodies at Mach numbers of 1·2, 1·4 and 1·6 and Reynolds numbers of 48 to 72 millions. The bodies consist of a basic forebody and parallel portion to which are added truncated parabolic afterbodies of three different thickness ratios. The calculations are in three main parts: (i) Calculation of the inviscid flow over the bodies, mainly by the method of charactertistics. (ii) Calculation of the boundary layer properties by an extension to compressible flows of the method of Squire and Young. (iii) Calculation of the pressure distribution on the "modified" afterbodies which result from adding the displacement thicknesses to the original profiles, by Ferri's method of linearised characteristic.—(1.1.3.4 × 1.6.1).

*A note on turbulent boundary-layer allowances in supersonic nozzle design. E. W. E. Rogers and B. M. Davis. C.P.*333. 1957.

Some of the theoretical methods for computing the growth of turbulent boundary layers along both the curved and straight walls of a rectangular supersonic wind tunnel are discussed and the estimated values compared with experiment. Approximate formulae, which may be useful in the initial stages of nozzle design, are suggested for the overall boundary layer growth along the curved nozzle and flat wall.—(1.1.3.4).

Aerofoils in cascade with boundary layer control. C. H. J. Johnson. ARL Note ME. 222. *July* 1957.

A design is presented of a cascade of aerofoils with boundary layer control obtained by using distributed suction.—(1.1.5 × 1.4.1).

Décollement laminaire en trois dimensions sur un obstacle fini. E. A. Eichelbrenner. O.N.E.R.A. Pub. No. 89. 1957.

Une définition intrinsèquement tridimensionnelle du décollement laminaire est donnée à l'aide de considérations géométriques et d'une hypothèse de régularité en fluide visqueux. Cette définition est appliquée au cas d'un ellipsoïde aplati avec ou sans incidence, placé dans un écoulement incompressible; cette dernière restriction n'est cependant pas essentielle.—(1.1.1.1).

Écoulement d'un fluide entre deux plans parallèles contribution a l'étude des butées d'air. R. Comolet. Pubs Sc. et Tech. 334. 1957.

L'auteur procède à une étude systématique des écoulements bidimensionnels d'un fluide visqueux en mouvement longitudinal ou radial entre deux plans parallèles fixes, la loi de transformation étant du type polytropique $(p/\rho^m)=C^{te}$. Cette étude présente une certaine généralité puisqu'elle tient compte des forces d'inertie dont les effets sont assez surprenants quand il s'agit d'écoulement radial divergent. Les formules obtenues permettent de retrouver quelques résultats déjà connus dans des cas plus simples. Des vérifications expérimentales complètent l'étude théorique et précisent le domaine d'application des formules proposées.—(1.1).

Beeinflussung der Lage des Verdichtungsstosses bei Profilen in hoher Geschwindigkeit. H. C. Sharaf. Max-Planck-Inst. Mitt. 16. 1957.—(1.1.3.3).

Contributions on the mechanics of boundary layer transition. G. B. Schubauer and P. S. Klebanoff. N.A.C.A. Report 1289. 1956.

The manner in which flow in a boundary layer becomes turbulent was investigated on a flat plate at wind speeds generally below 100 ft. per second. Hot-wire techniques were used, and many of the results are derived from oscillograms of velocity fluctuations in the transition region. The more familiar aspects of transition are discussed, and the facts discovered while studying the characteristics of artificially produced turbulent spots are presented.—(1.1.2.1).

Stability of laminar boundary layer near a stagnation point over an impermeable wall and a wall cooled by normal fluid injection. M. Morduchow et al. N.A.C.A. T.N. 4037. *August* 1957.

Minimum critical Reynolds numbers for laminar instability of two-dimensional stagnation flows over a wall cooled by normal fluid injection are determined theoretically. The individual as well as the net effects of wall temperature and normal fluid injection are investigated including the special case of zero injection rate. Comparison is made with results for flow over a flat plate.—(1.1.1 × 1.9).

A discussion of cone and flat-plate Reynolds numbers for equal ratios of the laminar shear to the shear caused by small velocity fluctuations in a laminar boundary layer. N. Tetervin. N.A.C.A. T.N. 4078. *August* 1957.

By use of the linear theory of boundary layer stability and Schlichting's formula for the maximum amplification of a disturbance, an approximate relation is derived between the Reynolds number on a cone and the Reynolds number on a flat plate for equal closeness to transition.—(1.1.2).

A thermal system for continuous monitoring of laminar and turbulent boundary layer flows during routine flight. N. R. Richardson and E. A. Horton. N.A.C.A. T.N. 4108. *September* 1957.

A thermal system has been developed which could be used to determine whether the boundary layer on a wing in flight is turbulent or laminar. Detectors cemented to the wing surface combine the functions of heating and temperature measurement. Detection is based on the difference in rate of heat transfer to a turbulent boundary layer as compared with that to a laminar boundary layer.—(1.1.2.1).

COMPRESSIBLE FLOW—*see also* LOADS

Measurements of the effect of windscreen shapes on the drag of cockpit canopies at transonic and low supersonic speeds using the free-flight-model technique. C. Kell. R. & M. 3024. 1957.

The effect if changes in the shape of the windscreen on the drag of a cockpit canopy has been measured on models in free flight. Canopies were attached to the models by means of a flexible mounting in such a way as to allow the canopy drag to be measured directly. The drag of the canopy was measured over a range of Mach numbers between 0·85 and 1·55 and the results show the improvements to be obtained by variations in the sweep of a vee windscreen and changes in the angle of a sloping windscreen.—(1.2.2 × 1.12.2).

The analysis of simple wave flow using a system of constant entropy shock waves. L. F. Henderson. A.R.L. Report ME. 84. *June* 1957.

This paper is concerned with developing a method for dealing with simple wave flow problems. The main assumption made is to regard the flow field Mach lines as shock waves of small intensity. In this way the shock wave equations sometimes lead to an easier method of solution than by the Method of Characteristics. The two-dimensional theory so developed is applied to a two-dimensional wing enabling calculated and measured pressure distributions to be compared.—(1.2.3.2).

A method for the asymptotic expansion of the integral equation of a lifting surface at near-sonic speeds. W. Eckhaus. N.L.L. T.N. F. 200. *April* 1957.

A new method for asymptotic expansion of singular integral equations is presented. The method can be applied to a large class of integral equations, of which the kernel contains a small parameter. In this report the integral equation of a lifting surface at Mach numbers near unity is chosen as an example for which the method is exposed.—(1.2.3.1).

Properties of test section walls with longitudinal slots in curved flow for subsonic and supersonic velocities (Theoretical investigations). B. H. Goethert. ASTIA AD-131406. August 1957.

Mathematical expressions for the boundary conditions of longitudinally slotted walls are derived in order to facilitate calculations to determine the effectiveness of slotted walls in wind tunnels. A relationship is first established for the effect of flow curvature on the boundary conditions in the vicinity of conventional slotted walls for incompressible flow. This relationship is then extended to include the influences of protruding slats as well as of perforated cover plates at the base of the slots. Finally the extent to which the derived equations are valid for compressible flow in the subsonic and supersonic speed range is shown.—(1.2.2.1 × 1.12.12).

The effect of blunting the nose of a flow angularity cone with 30° *included angle. R. J. Guthrie. ASTIA AD-131410. July* 1957.

Two nearly identical flow angularity cones with 30° included angle were calibrated at Mach numbers of 2·0, 3·1, and 3·5 over a range of angle of attack from +2·5 to −2·5°. The slope $\Delta p/p_{t_0}\alpha$ was experimentally determined for several configurations which differed in degree of bluntness of the cone. A flat 0·029 in. diameter on the nose on a cone with the orifice located 0·40 in. behind the apex was the most blunt configuration tested.—(1.2.3 × 1.12.5).

Nonlifting wing-body combinations with certain geometric restraints having minimum wave drag at low supersonic speeds. H. Lomax. N.A.C.A. Report 1297. 1957.

Several variational problems involving optimum wing and body combinations having minimum wave drags for different kinds of geometrical restraints are analysed. Particular attention is paid to the effect on the wave drag of shortening the fuselage and, for slender axially symmetric bodies, the effect of fixing the fuselage diameter at several points or even of fixing whole portions of its shape.—(1.2.3).

Bodies of revolution having minimum drag at high supersonic airspeeds. A. J. Eggers et al. N.A.C.A. Report 1306. 1957.

Newtonian impact theory is used to determine body shapes of minimum drag under various combinations of the conditions of given body length, base diameter, surface area, and volume. In addition an estimate is made of centrifugal forces, and their effect on the minimum-drag shapes is considered.—(1.2.3.1).

Equations, tables, and figures for use in the analysis of helium flow at supersonic and hypersonic speeds. J. N. Mueller. N.A.C.A. T.N. 4063. *September* 1957.

This report presents equations, tables, and figures for use in the analysis of helium (specific-heat ratio of 5/3) flow at supersonic and hypersonic speeds (Mach numbers 1 to 40). The equations provide the perfect-gas relations for continuous one-dimensional flow, normal- and oblique-shock waves, and Prandtl-Meyer expansions. The tables present useful dimensionless ratios for continuous one-dimensional flow and for normal-shock waves as functions of Mach number. Figures of the helium viscosity relation as a function of temperature, mass-flow rates as a function of Mach number and temperature, the Reynolds number as a function of Mach number and stagnation temperature, and the oblique-shock characteristics of wedges and cones at Mach numbers of 12, 16, 20, and 24 are shown.—(1.2.3).

Analysis of shock motion in ducts during disturbances in downstream pressure. H. G. Hurrell. N.A.C.A. T.N. 4090. *September* 1957.

A linearised analysis is made of the effect of small downstream pressure disturbances on the position of a normal shock in a duct with area variation. In the analysis, the gas flow is treated as quasi-one-dimensional and non-viscous. —(1.2.3.2 × 1.5.1).

CONTROL SURFACES—*see also* FLUID DYNAMICS

Some further jet flap experiments. N. A. Dimmock. C.P. 345. 1957.

The jet deflection is 58·1° from the chord line. In addition to the measurement of the forces and moments acting on the model the flow pattern in the vicinity of the aerofoil was revealed with smoke filaments and was recorded photographically.—(1.3.4 × 1.10.2.1).

Installation de la portance sur un profil muni d'une fente de soufflage au bord de fuite et correction due a l'envergure finie. R. Hirsch. Pubs. Sc. et Tech. N.T. 69. 1957.

Condition de pointe *AR* et équation intégrale de formation de circulation, homologue de celle de Wagner du problème sans jet. Résolution après calcul de la loi de freinage du jet et d'éloignement des éléments tourbillonnaires, effet dissipateur de Prandtl inclus. Extension à l'aile d'envergure finie et expressions des portances et traînée dans le cas limite stationnaire. Confrontations expérimentales.—(1.3.4 × 1.10).

Wind-tunnel investigation of external-flow jet-augmented double slotted flaps on a rectangular wing at an angle of attack of 0° *to high momentum coefficients. E. E. Davenport. N.A.C.A. T.N.* 4079. *September* 1957.

A wind tunnel investigation has been made to determine the characteristics of external-flow jet-augmented double slotted flaps which appear suitable for application to aeroplanes with pod-mounted engines. The investigation included tests of a rectangular wing with an aspect ratio of 6 over a momentum-coefficient range from 0 to 28.—(1.3.4).

Wind-tunnel investigation at low speeds to determine flow-field characteristics and ground influence on a model with jet-augmented flaps. R. D. Vogler and T. R. Turner. N.A.C.A. T.N. 4116. *September* 1957.

Results are presented of a wind tunnel investigation made at low speeds to determine the flow-field characteristics and ground influence on an aeroplane model having an untapered, unswept wing with an aspect ratio of 8·3 equipped with jet-augmented flaps deflected 55° and 85°. The jet-momentum coefficients covered a range representative of that of the output of current jet aeroplanes.—(1.3.4).

FLUID DYNAMICS—*see also* BOUNDARY LAYER

Preliminary analysis for a jet flap system in two-dimensional inviscid flow. E. C. Maskell and S. B. Gates. C.P. 359. 1957.

An attempt is made to clarify the background of the subject in two ways:—(1) By establishing some properties of a thin jet in a uniform field of flow. (2) By stating some momentum theorems connected with a jet issuing from a body in a uniform field of flow.—(1.4 × 1.3.4).

INTERNAL FLOW—*see also* COMPRESSIBLE FLOW

The secondary flow in a cascade of turbine blades. W. D. Armstrong. R. & M. 2979. 1957.—(1.5.4.3).

The design and testing of an axial compressor having a mean stage temperature rise of 30 *deg. C. A. D. S. Carter et al. R. & M.* 2985. 1957.

A compressor which was designed to give a mean stage temperature rise of 30°C is described. It has six stages so that the overall pressure ratio at the design point is 4·5 : 1. Full details of the factors which led to the form adopted and of the design itself are given.—(1.5.2.1).

A survey of the alternating pressures exciting high frequency vibrations in gas turbines. J. R. Forshaw. R. & M. 2989. 1957.

A survey of the predominant harmonic components of the alternating pressures is made for three centrifugal compressors, three axial-flow compressors and a single-stage turbine to ascertain the forces exciting vibration in a compressor or turbine stage.—(1.5 × 1.6.3).

*Dual-operation powerplants for vertical take-off transport aircraft: A preliminary analysis of several gas-turbine cycles. E. P. Cockshutt. N.A.E. Report LR-*201. *July* 1957.

Cycle calculations are presented for dual-operation gas-turbine power plants intended for use in VTOL aircraft. Cruising thrust is produced by expelling the working gas through propelling nozzles, and hovering lift is produced by expanding the working gas through turbines which in turn drive thrust augmentors (fans or propellers).—(1.5.3).

The principles of turbulent heat transfer. H. Reichardt. N.A.C.A. T.M. 1408. *September* 1957.

Relations are derived for calculating heat-transfer coefficients for turbulent and laminar fully developed flow and heat transfer in pipes and channels at constant wall temperature. Nusselt numbers for a round pipe are computed for Prandtl numbers from 0 to 1,000 at a Reynolds number of 30,000.—(1.5.1.1 × 34.3).

Aerodynamic forces on a vibrating unstaggered cascade. H. Söhngen. N.A.C.A. T.M. 1412. *August* 1957.

The unsteady aerodynamic forces, based on two-dimensional incompressible flow considerations, are determined for an unstaggered cascade, the blades of which are assumed to vibrate in phase with one another in an approach flow parallel to the blades.—(1.5.4.1 × 1.6.3).

LOADS—*see also* BOUNDARY LAYER
INTERNAL FLOW
WINGS AND AEROFOILS

Effects of elevator circuit stiffness on the loading conditions in longitudinal manoeuvres. D. R. Puttock. C.P. 354. 1957.

An investigation of the effects of elevator circuit stiffness on the loading conditions of an aircraft in a rapid pilot-induced pull-out manoeuvre is presented with a particular reference to the airworthiness aspects. The solution to the problem is given in general terms; it includes expressions for the incremental elevator angle, stick force, normal acceleration at the centre of gravity and the tailplane load in response to a chosen sequence of pilot-induced stick movements to initiate a rapid pull-out manoeuvre.—(1.6.2 × 1.8.2).

Slender-body calculations of local lift on some non-axially symmetric bodies with application to wind tunnel half models. M. E. Graham and Z. O. Bleviss. Douglas Report SM-22771. April 1957.

Slender-body theory is applied to the calculation of the local lift distribution on certain bodies. Bodies of a special cross section with a flat centre region are investigated by means of an electrical analogue and bodies with elliptical cross sections of varying eccentricity are studied analytically. These bodies represent an idealisation of a fuselage half model which is placed outside the wall boundary layer by using a shim between the model and the wall. The lift on such a half model is compared with half the lift on the full model.—(1.6.1 × 1.2.1).

STABILITY AND CONTROL—*see also* LOADS

A method utilizing data on the spiral, roll-subsidence, and Dutch roll modes for determining lateral stability derivatives from flight measurements. B. B. Klawans and J. A. White. N.A.C.A. T.N. 4066. *August* 1957.

A method for determining lateral stability derivatives from flight measurements is obtained by arranging the lateral equations of motion in such form that information from each of the three modes of lateral motion may be utilised.—(1.8.1.2 × 13.2).

Static longitudinal and lateral stability characteristics at low speed of 45° sweptback-midwing models having wings with an aspect ratio of 2, 4 or 6. D. F. Thomas and W. D. Wolhart. N.A.C.A. T.N. 4077. *September* 1957.

Results are presented of tests conducted in the Langley stability tunnel to determine the effects of various components and combinations of components on the static longitudinal and lateral stability characteristics at low speed of a series of 45° sweptback-midwing-aeroplane configurations having wings with an aspect ratio of 2, 4, or 6. The tests were made at a dynamic pressure of 24·9 lb./ft.^{2} which corresponds to a Mach number of 0·13 and Reynolds numbers of $1{\cdot}00 \times 10^6$, $0{\cdot}71 \times 10^6$, and $0{\cdot}58 \times 10^6$ based on the respective wing mean aerodynamic chords. The angle-of-attack range covered was from −4° to 32° and the sideslip angles used for the lateral-derivative tests were 5° and −5°.—(1.8.1.2 × 1.8.2.2).

Wind-tunnel investigation of effects of ground proximity and of split flaps on the lateral stability derivatives of a 60° delta-wing model oscillating in yaw. B. M. Jaquet. N.A.C.A. T.N. 4119. *September* 1957.

An investigation was made in the Langley stability tunnel to determine the effects of the proximity of the ground and of split flaps on the lateral stability derivatives of a 60° delta-wing model oscillating continuously in yaw. The model was tested with and without the vertical tail and the split flaps, without ground, and at ground-to-model distances from 0·30 to 1·25 wing mean aerodynamic chord at angles of attack to 30°. The test Mach number was 0·13 and the Reynolds number based on the wing mean aerodynamic chord was $1{\cdot}6 \times 10^6$.—(1.8.1.2 × 1.10.2.2).

THERMO-AERODYNAMICS—*see also* BOUNDARY LAYER

A simple model of the nonequlibrium dissociation of a gas in Couette and boundary layer flow. J. E. Broadwell. Douglas Report SM-22888. August 1957.

A simple model of a gas is formulated to illustrate the effect of the processes of dissociation and recombination on heat transfer in three situations: (1) In a layer of gas at rest between two walls at different temperature. (2) In Couette flow. (3) In a boundary layer on a flat plate. Two chemical conditions of the walls are considered: (*a*) The walls are catalytic. (The surface reaction rate is assumed sufficiently high to hold the gas at the wall in thermodynamic equilibrium). (*b*) The walls have no effect on the reaction; *i.e.* they are non-catalytic. A mechanical analogy to the stationary gas and Couette flow problems is developed. It provides, in these cases, a means of visualising the effect of the parameters and boundary conditions on the temperature and concentration profiles.—(1.9.1 × 1.1 × 34).

Influence of turbulence on transfer of heat from cylinders. J. Kestin and P. F. Maeder. N.A.C.A. T.N. 4018. *October* 1957.

This report deals with the problem of the influence of free-stream turbulence on the transfer of heat from a cylinder in forced convection at very low Mach numbers but at large Reynolds numbers. An attempt is made to determine whether the sole influence of turbulence is to shift the point of laminar separation in subcritical flow, or the point of transition in supercritical flow, and thus effect a change in the rate of heat transfer.—(1.9.1 × 1.1.3.1).

Experimental investigation of transpiration cooling for a turbulent boundary layer in subsonic flow using air as a coolant. W. E. Brunk. N.A.C.A. T.N. 4091. *October* 1957.

Tests were run for both constant coolant-flow rate and constant wall temperature along a flat porous plate.—(1.9.1 × 1.1.3.2).

Investigation of heat transfer from a stationary and rotating conical forebody. R. S. Ruggeri and J. P. Lewis. N.A.C.A. T.N. 4093. *October* 1957.

Experimental convective heat transfer was determined for a conical forebody (15° half-angle) for free-stream velocities up to 400 ft./second, rotational speeds up to 1200 r.p.m., angles of attack of 0° and 6°, and for heating conditions of uniform surface temperature and uniform heater input power density.—(1.9.1).

WINGS AND AEROFOILS—*see also* CONTROL SURFACES
STABILITY AND CONTROL
AEROELASTICITY
AIRCRAFT

Note on the circulation in circuits which cut the streamlines in the wake of an aerofoil at right-angles. J. H. Preston. R. & M. 2957. 1956.—(1.10.2.1).

A theoretical study of the aerodynamics of slender cruciform-wing arrangements and their wakes. J. R. Spreiter and A. H. Sacks. N.A.C.A. Report 1296. 1957.

A theoretical study is made of the pressures, loadings, forces and vortex wake associated with certain cruciform wing arrangements. For 45° bank, the wake of a cruciform wing is treated numerically with 40 vortices and analytically with 4 vortices. Comparisons are made with water-tank measurements, and the calculation of loads on cruciform tails by reverse flow is considered.—(1.10.1.2 × 1.6.1).

Investigation of the effects of profile shape on the aerodynamic and structural characteristics of thin, two-dimensional airfoils at supersonic speeds. E. D. Katzen et al. N.A.C.A. T.N. 4039. *September* 1957.

The effects of thickness, trailing-edge bluntness, boat-

tailing, and forward profile were determined. The results indicated that available theoretical methods were adequate for calculating the lift and pitching moment of the aerofoils under all conditions of the test. The theoretical methods for calculating foredrag and correlation curves for estimating base pressure were adequate for predicting the total drag when the position of transition was known.—(1.10.2.1).

The use of pure twist for drag reduction on arrow wings with subsonic leading edges. F. C. Grant. N.A.C.A. T.N. 4104. *August* 1957.

Linearised-theory calculations are presented for the drag reduction achieved by applying the first three terms of a power series for twist to flat delta wings and by applying linear twist to flat arrow wings with sonic trailing edges. Tip and root incidence angles for the optimum linearly twisted wings are shown. The calculations cover the speed range of subsonic leading edges.—(1.10.1.3).

TESTING AND INSTRUMENTS—*see also* COMPRESSIBLE FLOW

Experiments on Pitot-tubes in shear flow. F. A. MacMillan. R. & M. 3028. 1957.

Measurements were made in a pipe of circular section, with turbulent flow, using pitot-tubes of different external diameter. Some supporting experiments were also made in a turbulent boundary layer on a flat plate with zero pressure gradient.—(1.12).

Design and operation of hypersonic wind-tunnels. R. Smelt and J. C. Sivells. AGARD Report 135. *July* 1957.

Problems of the operation of hypersonic wind tunnels are discussed, including air supply, diffuser efficiency, heater design and operation, nozzle design and calibration, and instrumentation. Real (imperfect) gas effects and the application of hypersonic-wind-tunnel results to flight problems are also discussed. Some solutions are given, based on experience of the design and operation of the Gas Dynamics Facility at the Arnold Engineering Development Center.—(1.12.1.3).

On the influence of wall boundary layers in closed transonic test sections. S. B. Berndt. FFA. Report 71. 1957.—(1.12.1.2 × 1.1.3.3).

Theoretical aspects of the calibration of transonic test sections. S. B. Berndt. FFA. Report 74. 1957.

By applying the transonic similarity rule and the area rule to the flow in the test section a considerable reduction of the work involved in such a calibration appears to be possible for slender models. This is especially true for test sections having longitudinal slots of constant width, where the proper slot setting for zero interference at sonic speed is shown to be applicable to a large group of models of different size and shape (provided the influence of the wall boundary layers is negligible).—(1.12.1.3).

Wind-tunnel investigation of a number of total-pressure tubes at high angles of attack: subsonic, transonic and supersonic speeds. W. Gracey. N.A.C.A. Report 1303. 1957.

The total-pressure errors of 54 total-pressure tubes at angles of attack up to 60° and Mach numbers from 0·26 to 1·62 are presented. The configurations of the various tubes differed as regards external shape, internal shape, and type of total-pressure entry. External shapes tested included cylindrical, conical, and ogival nose sections. Internal shapes and total-pressure entries were varied on the basis of such factors as size of impact openings, internal chamber shape, leading-edge slant profile, and shielding (combined with venting). The effect of these variables on the sensitivities of the tubes to angle of attack is presented.—(1.12).

The measurement of pressure altitude on aircraft. W. Gracey. N.A.C.A. T.N. 4127. *October* 1957.

The various errors which determine the accuracy of an altimeter system are discussed and numerical values are assigned to each error. A statistical method for combining the individual errors is described. The overall errors of altimeter installations are calculated for Mach numbers up to 1·0 and altitudes up to 40,000 ft. An indication of the means by which some of the errors can be reduced is given. Various systems of barometric reference are also discussed.—(1.12.5 × 13).

AEROELASTICITY

Calculation of stability derivatives for oscillating wings. D. E. Lehrian. R. & M. 2922. 1956.

Derivatives are obtained for the following wings describing plunging (vertical translational) and pitching oscillations: (i) Delta wings of aspect ratio $A_r = 1{\cdot}2$, 2 and 3 with a taper ratio 1/7. (ii) Arrowhead wing of aspect ratio 1·32 with a taper ratio 7/18 and angle of sweep of 63·4° at quarter chord. (iii) Circular plate. Comparison is made with measured values of the derivatives for the delta wing of aspect ratio 1·2 and the arrowhead wing oscillating in incompressible flow.—(2 × 1.10).

Aerodynamic theory and its application to flutter. I. E. Garrick. AGARD Report 34. *April* 1956.

A survey is given of the present status of aerodynamic theory in relation to flutter analysis with a discussion of concepts implied in the structural and aerodynamical background. Applications of some recent developments in the theory of oscillating wings, offering promise for the flutter analysis of low-aspect-ratio wings, are indicated. Problems of the subsonic and supersonic speed ranges are included. —(2).

Divergence of delta and swept surfaces in the transonic and supersonic speed ranges. F. W. Diederich. AGARD Report 42. *April* 1956.

Various divergence phenomena are discussed in a qualitative manner. Methods of analysing divergence as well as subcritical static aeroelastic phenomena are discussed in some detail, with particular reference to approaches suitable for problems where simple-beam theory is inadequate. A simple approximate method of analysing static aeroelastic phenomena is presented and used to make some calculations for a delta wing at supersonic speeds.—(2).

A review of the present position on flutter. H. Templeton. AGARD Report 57. *April* 1956.

A summary of the present position on the flutter problem is given and recommendations made for further action.—(2).

Comparative calculations of divergence speed for wings of not too large aspect ratio. A. I. van de Vooren and O. Schatz. NLL-TN F 197. *January* 1957.

A comparison is made of divergence speeds calculated by aid of strip theory, lifting line theory and lifting surface theory. Due to the forward shift of the centre of pressure occurring for wings of low effective aspect ratio, the value found by lifting surface theory can be appreciably lower than that obtained from lifting line theory. Also the torsional deformations below the divergence speed will be underestimated by lifting line theory.—(2).

AIRCRAFT

A review and summary of the available aerodynamic data on deflected slipstream arrangements suitable for VTOL applications. J. A. Dunsby. N.A.E. Report LR-205. *September* 1957.

A review is made of the existing experimental and theoretical aerodynamic data on wing-propeller combinations intended for vertical take-off utilising the deflected slipstream principle. While a considerable volume of data is available indicating the trends of many important characteristics, it is clear that much theoretical and experimental work remains to be done.—(3).

Preliminary wind tunnel tests of a lifting fan in a two-dimensional aerofoil. R. L. Wardlaw and R. J. Templin. N.A.E. Report LR-207. *September* 1957.

Preliminary wind tunnel tests done with a two-dimensional aerofoil with a single fan mounted in the wing are outlined. A simple momentum theory is developed which provides reasonable agreement with the test results at low speeds. The use of fan outlet guide vanes to reduce drag is discussed, and the results of one test are presented.—(3 × 1.10.2.1).

American development in STOL and VTOL aircraft. C. W. Meshier. AGARD Report 86. *August* 1956.

To date most research activity on aircraft has been directed towards the development of basic configurations to achieve the required performance. This paper discusses the low

speed control problems of short take-off and vertical take-off aircraft, to which little effort has been applied until recently.—(3).

AIRCRAFT OPERATION

VTOL transport aircraft: A brief discussion of Canadian requirements and project study results for tilting-wing configurations with two propellers and four propellers. P. J. Pocock and A. J. Bowker. N.A.E. Report LR-204. September 1957.

A study is made of the tilt-wing configuration for two proposed missions: Payload 2000 lb. and 5000 lb., Range 500 miles, Hovering time 10 min. at 6000 ft., 95°F. The results are presented in some detail, including an estimate of direct operating costs showing that VTOL transport aircraft can be expected to have direct operating costs amounting to between 1·65 and 1·95 times the corresponding figure for conventional aircraft.—(5.1).

FLIGHT TESTING

See AERODYNAMICS—STABILITY AND CONTROL
—TESTING AND INSTRUMENTS
INSTRUMENTS AND EQUIPMENT

HYDRODYNAMICS

Investigation of high length/beam ratio seaplane hulls with high beam loadings. Hydrodynamic stability Part 14. *The effect of a tailored afterbody on stability and spray characteristics with test data on model J. D. M. Ridland. C.P.* 351. 1957.

The effects of a tailored afterbody on longitudinal stability, spray, directional stability and elevator effectiveness are deduced from the results of tests on two models of length/beam ratio 11, which were alike in every respect except that of afterbody shape; one afterbody was of standard form and the other was tailored.—(17.2).

Investigation of high length/beam ratio seaplane hulls with high beam loadings. Hydrodynamic stability Part 19. *The interaction of the effects of forebody warp, afterbody length and afterbody angle on longitudinal stability characteristics. J. K. Friswell. C.P.* 352. 1957.

The interaction of the effects of the different parameters concerned is considered. It is found that by a redefinition of stability it is possible to predict these interactions as far as the undisturbed lower limit is concerned, but that otherwise there seems to be no simple law governing them. Some broad generalisations are however possible.—(17.2).

Full scale measurement of impact loads on a large flying boat (Sunderland Mk. 5) *Part III—Data for impacts on main step. R. Parker. C.P.* 340. 1957.

The results of a series of full-scale impact tests were compared with an appropriate theory in a previous report. Discrepancies between this data and the theory were shown and this report presents complete time histories of a number of the actual measurements.—(17.2 × 33.1.2).

Full scale measurements of impact loads on a large flying boat (Sunderland Mk. 5) *Part IV—Data for impacts on the afterbody. R. Parker. C.P.* 341. 1957.

A comprehensive series of landing impact tests has been carried out in both calm and rough water on a Sunderland flying boat. Results are given for landings in calm water at attitudes where appreciable pressures occur on the afterbody and comprise data on total forces and local pressures on the afterbody.—(17.2 × 33.1.2).

Full scale measurements of impact loads on a large flying boat (Sunderland Mk. 5) *Part V—Results of rough water tests. R. Parker and J. K. Friswell. C.P.* 342. 1957.

The results of a number of take-offs and landings in various sea conditions are presented in the form of statistical data, both local pressures and overall accelerations being given.—(17.2 × 33.1.2).

INSTRUMENTS AND EQUIPMENT

Les telemesures françaises. J. Idrac. AGARD Report 33. *February* 1956.

Résumé des télémesures françaises actuellement en service. Description sommaire des quatre principales télémesures françaises—les télémesures O.N.E.R.A., S.F.E.N.A., Turck et D.E.F.A.—avec, en conclusion, des indications sur les possibilités d'emploi de ces instruments pour les essais en vol des avions, compte tenu des moyens de mesure utilisés jusqu'ici.—(18 × 13).

MATERIALS

The polarographic determination of tin in electrodeposited chromium. L. Wilson. A.R.L. Note MET. 8. *August* 1957.

A method has been developed for the determination of tin in chromium as deposited from a chromic acid-hydrofluoric acid electrolyte. The method has been found suitable for concentrations of tin in the range 0·0 - 0·1 per cent.—(21.6.1).

Mechanism of electropolishing of copper in orthophosphoric acid solutions. K. F. Lorking. A.R.L. Report MET. 24. *July* 1957.

Theories are proposed to explain the mechanism of the macro and micropolishing processes on copper anodes in orthophosphoric acid solutions.—(21.5).

Production problems of titanium and its alloys. E Swainson and R. L. P. Berry. AGARD Report 95. *April-May* 1957.

The problems of reducing the ore to a metal of sufficiently low oxygen content to have adequate ductility and alloying capabilities are discussed; melting in the absence of air and without contamination from furnace refractories and other sources; realisation of the full alloying potential of the metal; fabrication of ingots of high strength alloy compositions into the various semi-finished forms required; and reclamation of titanium scrap.—(21.2.2).

Aluminium alloys in aircraft structures. G. Meikle. AGARD Report 102. *April* 1957.

Compositions of three groups of aluminium alloys used in the U.K. are discussed in relation to alloys used in the U.S.A. The properties of these alloys at elevated temperatures are mentioned. Brief reference is made to a possible aluminium alloy for use at temperatures up to 200°C. Some difficulties experienced with stress-corrosion are mentioned.—(21.2.2).

Mechanical properties of glass-fabric honeycomb cores. Forest Products Laboratory Report 1861. *March* 1957.

The results of tests of commercially produced glass-fabric honeycomb cores are presented. Detailed descriptions of core materials and testing procedures are given. Relationships are given for various core properties in different directions. Design values of core properties are presented in tabular form and as stress-strain curves.—(21.3.6).

General survey of data on the reliability of metal-bonding adhesive processes. Forest Products Laboratory Report 1862. *May* 1957.

Results are presented of a survey made among a number of representative aircraft fabricators and adhesive manufacturers to obtain information on the reliability of adhesive-bonding processes for metals based on the variability of lap-joint strength data from tests for adhesive acceptance, laboratory control, adhesive storage and production control.—(21.3).

Cooperative investigation of relationship between static and fatigue properties of wrought N-155 alloy at elevated temperatures. N.A.C.A. Subcommittee on Power-Plant Materials. N.A.C.A. Report 1288. 1956.

Extensive data are given relating properties of *N*-155 alloy under static, combined static and dynamic, and completely reversed dynamic stress conditions at room temperatures, 1000°, 1200°, 1350° and 1500°F. Duplicate data were obtained from as many types of fatigue testing machines as could be arranged.—(21.2.2.1 × 31.2.2.2.1.8).

MATHEMATICS

Tables for the numerical determination of the Fourier transform of a function of time and the inverse Fourier transform of a function of frequency, with some applications to operational calculus methods. C. R. Huss and J. J. Donegan. N.A.C.A. T.N. 4073. *October* 1957.

A set of tables is presented which provides a means for

the numerical transfer of data between the time and frequency domain by use of the Fourier transform and inverse Fourier transform. Several applications to operational methods are discussed briefly.—(22).

METEOROLOGY

Impingement of cloud droplets on 36·5-percent-thick Joukowski airfoil at zero angle of attack and discussion of use as cloud measuring instrument in dye-tracer technique. R. J. Brun and D. E. Vogt. N.A.C.A. T.N. 4035. *September* 1957.

The trajectories of droplets in the air flowing past a 36·5 per cent thick Joukowski aerofoil at zero angle of attack were determined. The amount of water in droplet form impinging on the aerofoil, the area of droplet impingement, and the rate of droplet impingement per unit area on the aerofoil surface were calculated from the trajectories and cover a large range of flight and atmospheric conditions.—(24).

POWER PLANT

Effect of fluid-system parameters on starting flow in a liquid rocket. R. P. Krebs. N.A.C.A. T.N. 4034. *September* 1957.—(27.3).

PROPELLERS

Low-speed experimental investigation of the Magnus effect on various sections of a body of revolution with and without a propeller. M. J. Queijo and H. S. Fletcher. N.A.C.A. T.N. 4013. *August* 1957.

An experimental investigation has been made at low speed to determine the Magnus effect on various sections of a body of revolution of which several sections could be rotated either as individual units or in combinations. The investigation included the measurement of the Magnus effect on the body alone and also on the body with a three-blade propeller. The tests were made over an angle-of-sideslip range from $-5°$ to 30°, and at rotational speeds from 0 to 8000 r.p.m. with the propeller off and from 0 to 5000 r.p.m. with the propeller on.—(29.1).

FATIGUE

See also MATERIALS
THEORY AND ANALYSIS

Direct stress fatigue tests on Redux-bonded and riveted double strap joints in 10 *S.W.G. aluminium alloy sheet. S. Kelsy and J. B. Spooner. C.P.* 353. 1957.

Results are given of tensile fatigue tests on sheet specimens in DTD 610B, 546B and 687A with double strap Redux and riveted joints.—(31.2.3.2.4.2 × 31.2.3.2.2.2).

Fatigue characteristics of a riveted 24*S-T aluminium alloy wing. Part III. Test results. J. L. Kepert et al. A.R.L. Report SM.*248. *October* 1956.

Fatigue tests have been conducted on 90 P-51D Mustang wings during an investigation into the fatigue characteristics of a typical 24S-T riveted wing. A tabulation of the fatigue life, the type of failure and the load range is given for each specimen. In addition, results are given of investigations into crack propagation rate, local strain distribution, types of fatigue failure, effect of preload and the variation in structural flexibility.—(31.2.4.2 × 33.2.3.1).

SCIENCE—GENERAL

Zur Strahlungsphysik der Photonen-Strahlantriebe und Waffenstrahlen. E. Sänger. Forsch. Phys. Strahl. Mitt. No. 10. *June* 1957.—(32.2)

STRUCTURES

LOADS—*See* HYDRODYNAMICS

THEORY AND ANALYSIS—*See also* FATIGUE
TESTING

An experimental verification of the theoretical conclusions of R.A.E. Technical Note No. Structures 156 (*A.R.C., C.P. No.* 286) *"A constructional method for minimising the hazard of catastrophic failure in a pressure cabin" with further comments on its implications. D. Williams. C.P.* 357. 1957.

The results of experiments carried out on a Comet I cabin that was modified to incorporate the new constructional method are recorded. These experiments amply confirm the benefits indicated by theory, and appear fully to vindicate the new method of construction.—(33.2.3 × 31.2.4.4).

Small deflection theory of flat plates using complex variables. Part 1. *Fundamental equations in complex form. P. D. Jones. A.R.L. Report SM.*252. *June* 1957.

The fundamental equations of the small deflection theory of flat plates are expressed in complex form and general solutions obtained for a variety of loading conditions. These include continuously differentiable pressure variations, point loads, line loads and regional loads.—(33.2.4.5.3).

*Edge stresses in thin shells of revolution. W. H. Wittrick. A.R.L. Report SM.*253. *June* 1957.

Explicit solutions are obtained for stress systems which decay rapidly with the distance from either a circular or a meridianal edge of a thin shell whose shape is a surface revolution, but otherwise arbitrary. The accuracy of the solutions is consistent with that of the generally accepted equations of thin shell theory. The significance of these solutions from the point of view of the boundary conditions of membrane theory is discussed.—(33.2.4.8.0).

*Graphical and tabulated data on the frequency and modal characteristics of swept cantilevers. A. H. Hall et al. N.A.E. Report LR-*193. *June* 1957.

This report relates to the vibration of swept cantilevers with parallelogram planform, and contains comprehensive data on the frequency and modal characteristics.—33.2.3.2 × 33.2.4.1.10).

Effets thermiques dans le calcul de la resistance des structures d'avions et d'engins. N. J. Hoff. AGARD Report 52. *January* 1956.

After reviewing the laws controlling the aerodynamic heating of supersonic aircraft and missiles, heat transmission inside the wings by conduction and radiation are discussed. Thermal stresses and their effect on the buckling of rods and thin plates are examined, as well as the problem of creep and its effect on stress distribution. The last part deals with buckling caused by creep.—(33.2.4.0.9).

Some remarks upon the problem of temperature shock in aircraft. F. Bollenrath. AGARD Report 90. *August* 1956.

Structural problems associated with heating and large temperature gradients are discussed. Thermal stresses due to a non-stationary distribution of temperature are calculated for a plate cooled on one surface and heated on the other. Formulae are obtained which make a comparison of different materials possible.—(33.2.4.5.9).

Application of obliquely mounted strain gage to measurement of residual stresses in disks. M. H. Hirschberg et al. N.A.C.A. T.N. 4027. *September* 1957.

A simple method is presented whereby the residual stress distribution in a disc can be determined. The residual tangential stresses are measured with the aid of the obliquely mounted strain gauge when the disc is cut in the recommended manner. Results of measurements on overspeeded discs compare favourably with the calculated values.—(33.2.4.0 × 33.3.2).

TESTING—*see also* THEORY AND ANAYLSIS

Theory of aircraft structural models subject to aerodynamic heating and external loads. W. J. O'Sullivan. N.A.C.A. T.N. 4115. *September* 1957.

The adequacy of an aircraft structure to withstand simultaneously the external loads and aerodynamic heating encountered in supersonic flight is examined by means of a structural model. By dimensional analysis it is shown that a structural model can be built which has the required similitude with respect to flow of heat through it, thermal stresses and deformations, and stresses and deformations due to external loads.—(33.3.2 × 33.2.3.0).

APPOINTMENTS

This section of **THE JOURNAL** *is available for advertisements of appointments in the Industry, the Ministries, Research Establishments, Universities and Colleges in the United Kingdom only.*

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CANCELLATION OF LECTURE

Owing to unforeseen circumstances the Section Lecture which was to have been given on 4th February by Mr. G. Selwood on "Power Controls for Supersonic Aircraft" has had to be cancelled.

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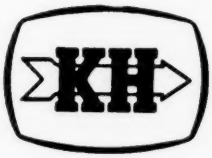

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